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Flying Qualities of Relaxed Static Stability Aircraft - Volume II

Ramifications of Flight-Essential/Critical Heavily-Augmented Airplane Characteristics on Flying Qualities

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Final Report

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16. Abstract

Volume I of the report deals with airworthiness assessment and flying qualities evaluation of highly augmented aircraft covered by Parts 23 and 25 of the Federal Aviation Regulations. Particular emphasis has been placed on aircraft with relaxed static stability and on the use of active augmentation systems to achieve the minimum requirement for a level of safety in such aircraft. Significant modifications and expansion to the FAA Engineering Flight Test Guides are detailed.

Volume II supports the work of Volume I and provides the more analytically oriented research results of this report. Emphasis is placed on determining the relative similarities and differences between heavily augmented and conventional aircraft. A number of important generic distinctions have been found and are described and explained.

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Remove the following pages and replace with (new) attached pages:

Volume II - Page 27 (Figure 9)
Volume II - Page 38 (Figure 14)
Volume II - Page 61 (Figure 18).

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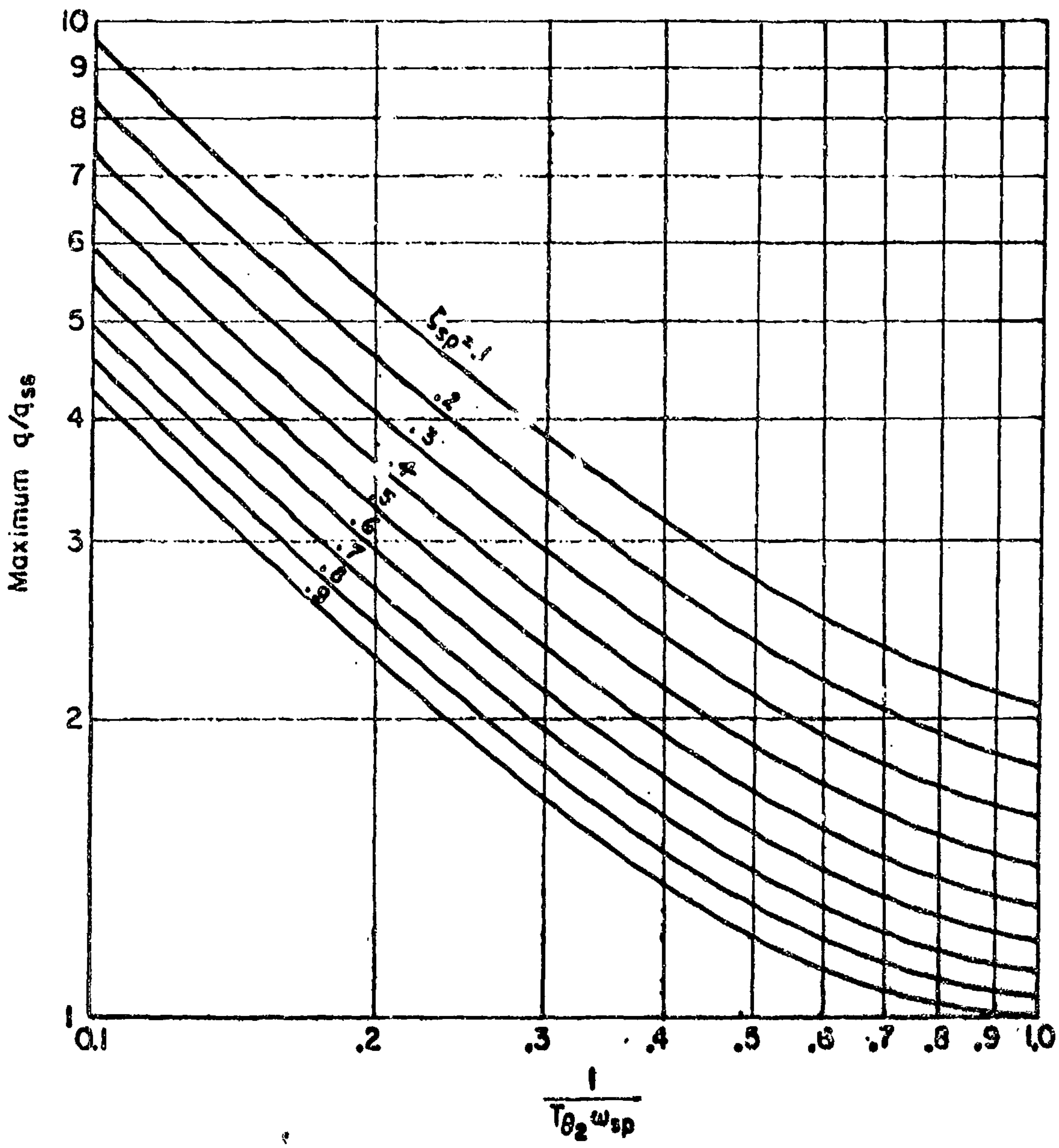

Figure 9. Maximum Pitch Rate Overshoot for Step Control Input

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control of the outer path deviation loop are, as already remarked, limited by the path/attitude lag, T_{θ_2}. For conventional airplanes, the augmented aircraft pitch dynamics lead has the same time constant, T_{θ_2}. (In other words, the $1/T$ in the augmented aircraft pitch dynamics in Figure 5 is $1/T_{\theta_2}$.) Thus the attitude lead and the path lag are the same quantity, fixed by the same aircraft configuration feature (the lift curve slope). The undamped natural frequency and damping ratio of the augmented aircraft pitch dynamics are, in this case, $\zeta = \zeta_{sp}$ and $\omega_n = \omega_{sp}$. Consequently, for this conventional aircraft the pitch and path dynamics are predominantly dependent on these three variables, T_{θ_2}, ζ_{sp}, and ω_{sp}, which in turn depend on the lift curve slope, the weathercock stability, and the pitch damping.

The variation of the aircraft short-period characteristics as static stability is reduced can easily be studied using a root locus approach. The idea is to examine root plots, such as Figure 8a, as M_α is varied. This is done by plotting the short-period roots for a number of given values of M_α, and then connecting the roots to form a locus.

The short-period characteristic function is given approximately by:

$$s^2 + 2(\zeta\omega)_{sp}s + \omega_{sp}^2 = s^2 - \left(Z_w + M_{\dot\alpha} + M_q\right)s + \left(Z_wM_q - M_\alpha\right)$$

$$\doteq \left(s - Z_w\right)\left(s - M_q\right) - M_\alpha$$

When the static stability is zero, i.e., the c.g. is at the neutral point and $M_\alpha = 0$, the short-period characteristic function reduces to $(s - Z_w)(s - M_q)$. These roots are used as starting points for the locus. The effects of M_α variation on the short period are shown as root and corresponding step function control input time response plots in Figure 10.

Figure 10a illustrates the root variation as M_α is permitted to become more negative (e.g., c.g. increasingly moved further forward of the neutral point). The short-period changes from the two roots (Z_w and M_q) at B to a rendezvous point at $(Z_w + M_q)/2$, and then break away to

from the phugoid, approach each other, rendezvous, and become a quadratic pair. As the static stability is further decreased, this pair approaches the classical two-degree-of-freedom phugoid mode wherein $\omega_p \doteq \sqrt{gZ_u/U_o} \doteq \sqrt{2g/U_o}$. In this classic phugoid (Reference 22) the angle of attack is fixed while airspeed and pitch attitude oscillate, interchanging potential and kinetic energy, damped only by drag. Thus, the actual divergence in the three-degree-of-freedom case does not stem from the short period, but rather from the phugoid. The important point to be noted is that stability is just neutral in the three-degree-of-freedom motions when the static margin is reduced to zero. Use of the short-period approximation indicates neutral stability when the maneuver margin is zero. (In all of this discussion, the pitching moment change due to speed change, M_u, is assumed to be zero, so that static stability is governed entirely by M_α.)

Another interesting perspective about the short-period response can be gained using the peak q/q_{ss} versus $1/T_{\theta_2}\omega_{sp}$ coordinates of Figure 9 as a backdrop for variations in stability. In principle it might seem that almost any response is available (i.e., any point within the $0 < \zeta < 1$ space of Figure 9 could be reached) if one only designs and balances the aircraft configuration properly. This is partly true in that any desired short-period damping ratio, ζ_{sp} can be achieved using a pitch damping augmentor. But, because the path/attitude lag is a given for a particular wing configuration, the adjustment of c.g. (and hence of M_α) can lead only to a tightly constrained set of dynamic response properties. This is shown in Figure [illegible] for the Generic RSS aircraft in cruise. The curve shows what is attainable in terms of overshoot, damping ratio, etc. For instance, it indicates that very large static margins are accompanied by large overshoots induced by both the ζ_{sp} and $1/T_{\theta_2} - \omega_{sp}$ spread. This is to be expected when total short-period damping is constant and ω_{sp} is increased, as occurs when the c.g. is moved forward (see Figure 13a). As static stability is reduced due to decreasing M_α as the c.g. is moved aft, the pitch rate overshoot decreases and the damping ratio increases. The curves defining the attainable dynamic properties can be shifted, mainly up and down, by

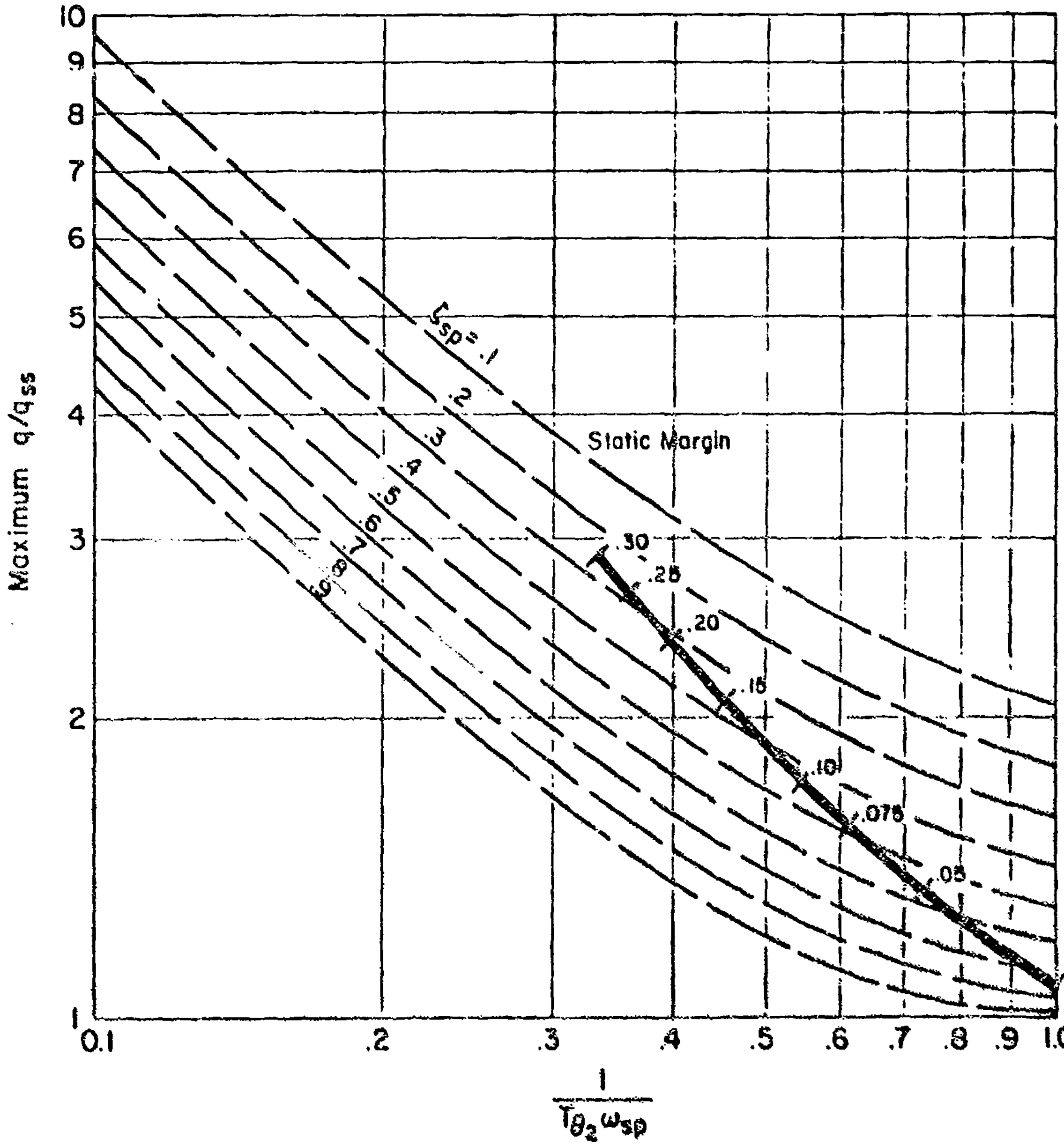

Figure 14. Pitch Overshoot Variation with Static Margin for Conventional Aircraft (Generic Aircraft in Cruise)

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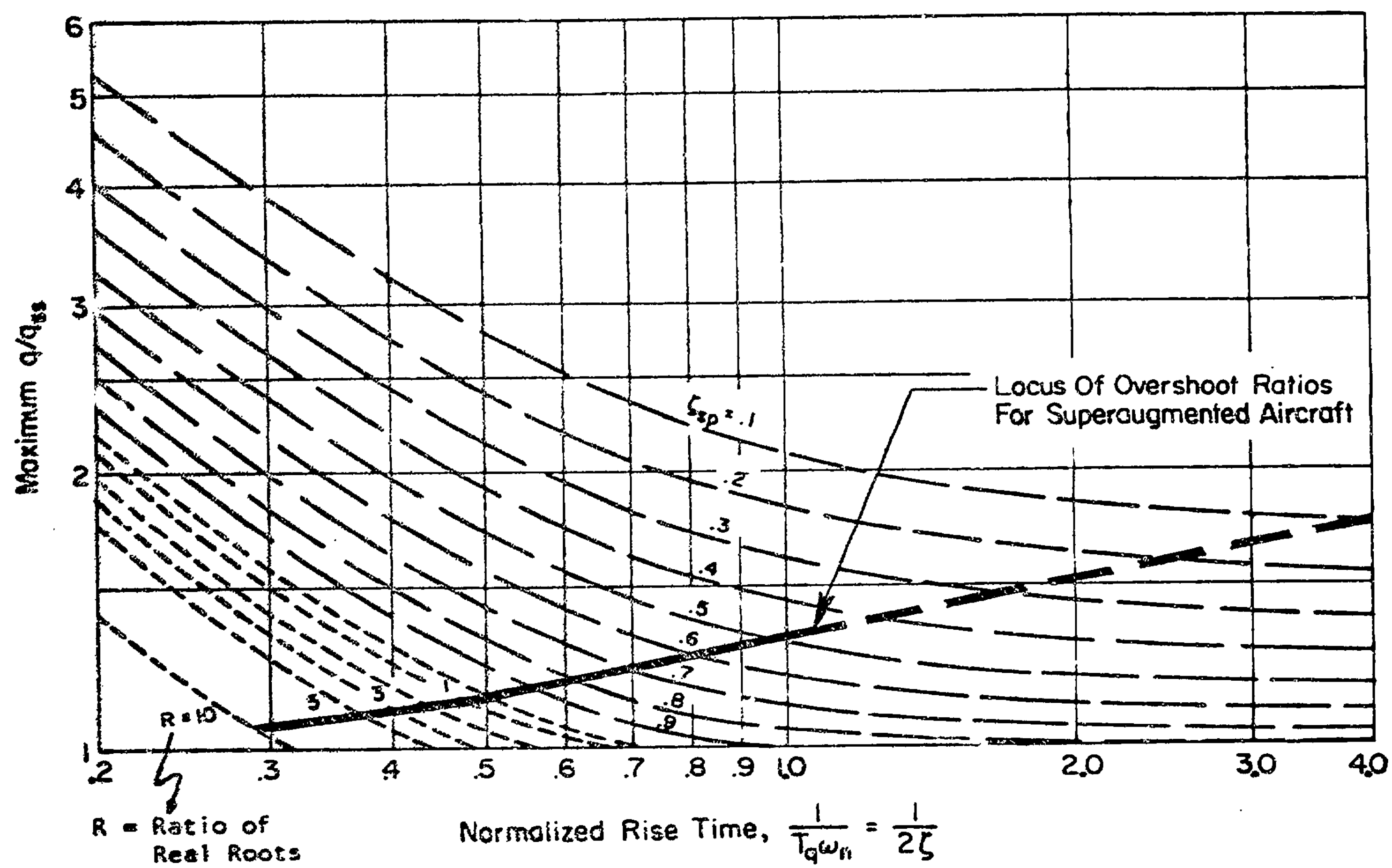

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Figure 18. Maximum Pitch Rate Over-hoot Variation for Superaugmented Aircraft

A major distinction can also be made between the superaugmented and conventional aircraft with reference to the aerodynamic characteristics which underlie their responses. For the conventional aircraft, even in the short period, the stability derivatives Z_w, M_q, and M_α together with their variations with flight condition, are major governing parameters. When the complete three-degree-of-freedom airplane characteristics are also taken into account, several more derivatives become important (e.g., Z_u, M_u, X_w, etc.). On the other hand, to the extent that the augmentation system can be made to approach the superaugmented characteristics, the aerodynamic parameters of importance reduce to the surface effectiveness, M_δ. Potential variations in other derivatives must, of course, be assessed in the design process to assure that no possible variation could upset this applecart, but in actual system operation the primary sensitivity and variations of interest are those of M_δ. In some ways, this sparsity of airplane-characteristic-dependence for aircraft which approach the superaugmented state offers a major advantage. The system which provides superaugmentation will itself be complex in that it is multiply redundant, yet the properties of any single channel of the multiple redundant system are extremely simple, straightforward, and sensitive to only a very few parameters. Thus, the concept of a "simplex" multiple redundant augmentor has some appeal and bears further consideration.

Finally, the ultimate comparison of the conventional and superaugmented vehicles is connected with the closed-loop precision path control flying quality aspects. Referring to Figure 5, we can now indicate why the augmented aircraft pitch dynamics block was not made more specific in terms of the subscripts for the quantities in the transfer function incorporated there. The attitude lead is now no longer T_{θ_2}, but the control system lead T_q, while the undamped natural frequency and damping ratio are unrelated to those of the conventional short period. Thus, the augmented aircraft pitch attitude dynamics are potentially fundamentally different than those of a conventionally augmented aircraft. Not the least important of these differences is the replacement of the T_{θ_2} lead by T_q, for now the attitude lead is not the same as the path/

PREFACE

The research reported herein was accomplished under Contract DTFA-03-81-C-00069 for the Department of Transportation, Federal Aviation Administration Technical Center at Atlantic City Airport, New Jersey. The contracting officer's technical representative (COTR) was Mr. Joseph J. Traybar (ACT-340), Flight Safety Research Branch, Aircraft Safety Development Division.

The authors wish to express their gratitude to Mr. Joseph Traybar for his many helpful comments and guidance during the performance of this work as well as for his considerable contributions during the review of this report.

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NOMENCLATURE

c	Reference chord length
C_D	Drag coefficient; $\frac{2D}{\rho U^2 S}$
C_{D_u}	$\frac{U}{2}\frac{\partial C_D}{\partial U}$
C_{D_α}	$\frac{\partial C_D}{\partial \alpha}$
C_{D_δ}	$\frac{\partial C_D}{\partial \delta}$
C_L	Lift coefficient; $\frac{2L}{\rho U^2 S}$
C_{L_u}	$\frac{U}{2}\frac{\partial C_L}{\partial U}$
C_{L_α}	$\frac{\partial C_L}{\partial \alpha}$
$C_{L_{\dot\alpha}}$	$\frac{\partial C_L}{\partial (\dot\alpha c/2U)}$
C_{L_q}	$\frac{\partial C_L}{\partial (qc/2U)}$
C_{L_δ}	$\frac{\partial C_L}{\partial \delta}$
C_M	Pitching moment coefficient; $\frac{2M}{\rho U^2 Sc}$

C_{M_u}	$\frac{U}{2}\frac{\partial C_M}{\partial U}$
C_{M_α}	$\frac{\partial C_M}{\partial \alpha}$
$C_{M_{\dot{\alpha}}}$	$\frac{\partial C_M}{\partial(\dot{\alpha}c/2U)}$
C_{M_q}	$\frac{\partial C_M}{\partial(qc/2U)}$
C_{M_δ}	$\frac{\partial C_M}{\partial \delta}$
D	Aerodynamic drag
g	Acceleration due to gravity (32.2 ft/sec^2)
h	Path deviation; altitude
h_c	Desired path
h_e	Path error
I_y	Moment of inertia in pitch
K	Gain
K_q	Controller gain
L	Aerodynamic lift
M	Aerodynamic pitching moment
M_q	(Single degree of freedom) pitch damping; Pitching acceleration per unit pitching velocity; $M_q = \frac{\rho S U c^2}{4 I_y} C_{M_q}$
M_u	Pitching acceleration per unit forward velocity; $M_u = \frac{\rho S U c}{I_y}(C_M + C_{M_u})$

M_w, (M_α)	Weathercock stability; Pitching acceleration per unit vertical velocity (angle of attack); $M_\alpha = U_0 M_w \;;\; M_w = \frac{\rho S U c}{2 I_y} C_{m_\alpha}$
$M_{\dot{\alpha}}$	Pitching acceleration per unit angle of attack rate $M_{\dot{\alpha}} = \frac{\rho S U c^2}{4 I_y} C_{M_{\dot{\alpha}}}$
M_δ	Pitching acceleration per unit control surface deflection $M_\delta = \frac{\rho S U^2 c}{2 I_y} C_{M_\delta}$
q	Pitching velocity
q_e	Pitching velocity error
q_{ss}	Steady-state pitching velocity
S	Reference planform area
s	Laplace Operator
T	Time constant
T_q	Lead time constant in augmentation system
T_r	Rise time
T_{θ_2}	Lead time constant in short-period θ/δ transfer function for conventional airplane dynamics. Lag time constant between flight path angle, γ, and pitch attitude, θ. $1/T_{\theta_2} = -Z_w + Z_\delta M_w / M_\delta \doteq -Z_w$
U	Speed
U_0	Speed in equilibrium condition
w	Vertical velocity
W	Gross weight
w_g	Vertical gust velocity

X_u	Forward acceleration per unit speed change $$X_u = \frac{\rho SU}{m}(-C_D - C_{D_u})$$
X_w, X_α	Forward acceleration per unit vertical velocity $$X_\alpha = U_o X_w \quad ; \quad X_w = \frac{\rho SU}{2m}(C_L - C_{D_\alpha})$$
Y_{p_h}	Pilot describing function for pilot control action on path deviation
$Y_{p\theta}$	Pilot describing function for pilot control action on attitude
Z_u	Vertical acceleration per unit forward velocity $$Z_u = \frac{\rho SU}{m}\left(-C_L - C_{L_u}\right)$$
Z_w, $(Z\alpha)$	Heave damping; vertical acceleration per unit vertical velocity (angle of attack); $$Z_\alpha = U_o Z_w \quad ; \quad Z_w = \frac{\rho SU}{2m}\left(-C_{L_\alpha} - C_D\right)$$
Z_δ	Vertical acceleration per unit control surface deflection $$Z_\delta = \frac{\rho SU^2}{2m}\left(-C_{L_\delta}\right)$$
α	Angle of attack
α_A	Aerodynamic angle of attack, $(w - w_g)/U_o$
α_I	Inertial angle of attack, w/U_o
δ	Generic control surface deflection
δ	Control Surface (elevator or horizontal tail) deflection
δ_{e_p}	Pilot command input
δ_H	Horizontal tail surface deflection (generic airframe)
ζ	Damping ratio
ζ_{sp}	Short period damping ratio
θ	Pitch attitude
$\dot{\theta}$	Pitch attitude rate

θ_c	Attitude Command
θ_e	Attitude Error
ρ	Atmospheric Density
σ	Real component of complex variable, s
τ_d	Delay time
τ_M	Delay margin, ϕ_M/ω_c
Φ, ϕ	Bank Angle
ϕ_M	Phase Margin
ω	Imaginary component of complex variable, s
ω_c	Crossover Frequency
ω_{c_a}	Crossover frequency of amplitude ratio asymptote

EXECUTIVE SUMMARY

Considerable attention has been given recently to the use of certain advanced aircraft configurations and flight control designs and the implementation of new system concepts in order to improve or optimize aircraft designs, flight characteristics, performance, and efficiency. Utilization of these new aircraft and system concepts to achieve these desired goals usually requires consideration of beneficial design factors (such as aft center of gravity and smaller sized tail-planes and empennage) that tend to cause poorer aircraft flying qualities characteristics for certain modes of flight. Therefore, for many new generation aircraft, it will be necessary to provide various tiers of stability and control augmentation to optimize the designs as well as compensate for potential problems associated with flying qualities safety requirements for failed-state conditions. The trend of using highly augmented flight systems is well established and indeed, in the recent NASA sponsored study for Energy Efficient Transports, the Boeing, Douglas, and Lockheed aircraft companies all recommend highly augmented airplanes for their proposed designs.

In the present study, the flying qualities of highly augmented aircraft are examined in the context of the current Federal Aviation Regulations (FAR) and supporting Engineering Flight Test Guides to determine if they require modification and/or updating. Also, attention is directed toward the determination of the data and information needed to adequately and efficiently assess the flying qualities airworthiness of highly augmented aircraft and systems to ensure that they meet the minimum requirements for a level of safety.

First, it must be clearly understood that the current flying qualities related FAR are based essentially on classical stability and control of unaugmented aircraft. Therefore, it was necessary to determine what specific differences exist between the flying qualities of classical unaugmented aircraft and the highly augmented (or super augmented) aircraft being proposed for greater performance and fuel efficiency. It has been the purpose of this study to make such determinations and updating of pertinent agency documents. The results of the study are presented in two volumes. Volume I contains a detailed review of the assessments as defined in the FAR and Engineering Flight Test Guides. Volume II contains a more detailed technical analysis of highly augmented and super augmented aircraft to provide analytical support for Volume I. The emphasis is on the longitudinal axis in keeping with the desire to provide fuel efficiency via relaxed static stability. However, some considerations of the lateral and directional axes have also been reviewed.

The difficulty in changing established regulations has been an overriding consideration, and suggestions to modify an existing FAR were made only when no alternative could be identified. In nearly all cases, the existing FAR have been found to be adequate with the important proviso that detailed interpretations and flight test procedures can be developed for inclusion in the supporting Engineering Flight Test Guide. However, it appears that the current versions of the Engineering Flight Test Guide do not provide adequate guidance to support the flying qualities airworthiness assessment of highly augmented aircraft and will require significant modifications and updating. In fact, many important sections in the Engineering Flight Test Guides are blank or missing and listed simply as "Reserved."

Specific areas of interest or possible activities needed to aid in upgrading the pertinent documents are detailed. Brief comments on some of these areas are: A synopsis of FAA pertinent data and information taken from applicable portions of flight test and simulations studies (as accomplished by NASA, DOD in the form of

reports and handbooks; e.g., MIL-F-8785C), should be culled and portions included in the FAA Engineering Flight Test Guides in a format that is readily usable to the agency and certification team members in the flying qualities airworthiness assessment process for minimum requirements for a level of safety. Specific piloting tasks should be defined for evaluation of critical aspects of certain features of highly augmented aircraft. Issues related to "long-term" dynamic stability requirements need to be fully and efficiently addressed. The idiosyncrasies of specific augmentation schemes should be discussed in some detail so FAA flight test engineers and test pilots can fully and efficiently evaluate such systems. For example, both active and passive augmentation schemes should be covered ranging from downsprings and bobweights all the way to highly redundant full-authority high-gain fly-by-wire systems. All aspects of augmentation system failures should be considered. For example, the Engineering Flight Test Guide should contain a clear interpretation of what constitutes "non-essential," "essential," and "critical" flight control functions. In addition, the effects of failure transients and critical conditions for failures should be spelled out in detail.

Currently, the minimum requirements for a level of safety are defined by several key phrases scattered throughout the FAR. For example, "without exceptional piloting skill, alertness, (attention) or strength" is the phrase used to distinguish between what is and what is not an acceptable level of safety in some paragraphs. A more definitive rating rationale and structure should be designed and considered for agency use by the FAA flight test pilots and engineers as an additional aid in determining more precisely what constitutes PASS/FAIL rating and compliance with "key-phrase" use for the evaluation of flying qualities minimum requirements for a level of safety. To this end, Volume I suggests that the well-known and widely used Cooper-Harper Rating Scale be more formally utilized, in truncated form, in the flying qualities appraisal process. That is, the Cooper-Harper "Rating" column, the Aircraft Characteristics column, and Demands on the Pilot (Workload) column are identically retained but the block diagram schematics for Adequacy for Selected Task have been excised and a new PASS/FAIL Judgment column (designed and calibrated specifically for FAA application) has been juxtaposed with the familiar 10-point rating scale for agency use in conjunction with the existing "key-phrases" of the FAR and flight test guides. This truncated version of the Cooper-Harper Flying Qualities rating system is offered here to reduce agency application difficulties and other past rating complexity issues. It provides an initial rationale with a more solid data foundation that should aid greatly in structuring all airworthiness PASS/FAIL appraisals. Also, use of this type of rating scale should eliminate or at least mitigate objections by some applicants related to relative rating comparisons (of "goodness" or "badness") of aircraft, systems, and products.

In the present study, we have defined specific areas of concern in aircraft flying qualities related Federal Aviation Regulations and associated Engineering Flight Test Guides when utilized for the airworthiness assessment of highly augmented aircraft.

SECTION I

INTRODUCTION

A great deal of attention has recently been given to the technological ways and means of improving aircraft energy efficiency and performance. It is believed that this trend will continue with, if anything, increasing urgency in the years ahead. A leading approach has been the development of new control concepts which can be used in conjunction with aircraft configuration tailoring to achieve a more efficient system product. Conservative versions of modern multiple fail operational flight control, as exemplified by the Space Shuttle and new fighters coming into the inventory, provide the capability for major weight and volume reductions and performance enhancements in the primary flight control system elements and for optimization of the basic airframe for performance properties such as low drag, longer fatigue life, etc. The new control technology permits this to be accomplished without some of the traditional penalties imposed by stability and control or flying quality requirements. Indeed, the new flight control technology can redress stability and control imbalances which earlier would have been considered to be excessive, while at the same time provide flying quality characteristics which border on the absolute optimum.

Because of these technological trends, it is important to review whether the existing Federal Aviation Regulations (FAR's) and their associated flight test guides are still pertinent to cope with potential advanced systems. In order to do this, the characteristics of representative advanced systems need to be compared with those of existing conventional aircraft to determine whether a procedure based on evolutionary, by analogy, extensions of existing knowledge and approaches is sufficient to cope with the future or whether some fundamental changes may be required. Accordingly, the primary purpose of this study is to determine the ramifications of relaxed static stability (RSS) airplane characteristics and associated advanced flight control system behavior on flying qualities, in general, but with a focus on aircraft covered by Part 25 of the FARs. In this report, the emphasis is on determining the

relative similarities and differences between heavily augmented and conventional aircraft. A number of important distinctions have been found and are described and explained here. Unfortunately, while distinctions have been drawn and issues developed, the existing data base from analysis, simulation and flight is insufficient to resolve some of the key questions. In this sense, this report can serve only as an introduction to a future agenda.

The approach we have taken is initially to review (in Section II) the growth of stability augmentation and its potential for taking care of aircraft dynamic deficiencies which may be incidental to aircraft configuration tailoring for energy efficiency. This amounts to an outline of incentives for heavy augmentation, i.e., for full authority, multiple redundant command augmentation systems. With such systems, the consequences of failure demand major attention as an integral part of the system evolution and operation.

We next (Section III) address the flying qualities and dynamics of relaxed static stability aircraft. Flying qualities are first divided into different categories of control functions needed to conduct flight operations, and associated levels of pilot involvement. The potential expansion of operating conditions defined by the FARs to include configurations associated with stability augmentation system (SAS) failures and allowable SAS-off aircraft dynamics follows. Attention is then focused on high workload pilot/aircraft control operations as the central issue in safety-related flying qualities. With this orientation precision path control is put forth as the key control task to explore the effects of heavy augmentation on closed-loop pilot/aircraft system flying qualities.

With precision path control as the focal point, the dynamic characteristics of a conventional aircraft are then developed and described. These are then compared with the similar dynamics of a relaxed static stability aircraft which is further illustrated using specific examples based on a set of generic RSS aircraft characteristics (provided in Appendix A).

The dynamics of the relaxed static stability aircraft become more unfavorable the greater the "relaxation". The greatest energy efficiency and performance benefits are gained when the static stability can sometimes be somewhat unstable and it is in these cases that heavy augmentation is required. Partial benefits can, of course, be obtained with relaxed static stability at a level wherein the aircraft-alone is still statically stable, but these conditions do not require full authority, heavy augmentation. To correct the aircraft-alone stability deficiencies, a number of augmentation schemes are possible. These are described, and then a particular type of system is put forth as an example. The characteristics of the aircraft/augmentation system are then developed and contrasted with those of the conventional aircraft. The discussion shows that a properly designed augmentor can result in attitude response characteristics which are identical in form to those of the conventional aircraft short period mode. On the other hand, the parameters which govern this response are entirely different in their origin and may be significantly different in their quantitative values. This feature of heavily augmented aircraft brings new flying quality considerations to the fore.

The next section (IV) starts with a summary of the fundamental flying quality differences between heavily augmented and conventional aircraft. It then explores these differences for their consequences viewed in the light of existing data. An exemplary set of response boundaries for RSS aircraft are used in this exploration as a "strawman" backdrop for comparing pertinent conventional and highly augmented system data. The example heavily augmented RSS aircraft used as our study example happens to fit well within these exemplary framework, as do some data for conventional aircraft. Interestingly enough, the conventional data which fall within the framework are for aircraft with very marginal flying quality characteristics. On the other hand, data for other conventional aircraft which do possess good flying qualities do not fall within the exemplary boundaries. One interpretation of this is that criteria which may apply as reasonable bounds on highly augmented aircraft are not sufficient to define a good conventional airplane. Then,

by analogy, the highly augmented airplane, which does fly within its strawman bounds, may have flying qualities which are unfavorable. On the other hand, the difference in characteristics between highly augmented and conventional aircraft may be responsible for the disparity. That this is a likely possibility is given credence by a datum which approaches the idealized heavily augmented condition (called superaugmented herein). For this particular simulated aircraft, the flying qualities were rated as high as a 1 by one pilot.

Other differences between heavily augmented and conventional aircraft are then explored. These include the pilot stick force variation with speed and many of the side effects of a particular mechanization on the flying qualities of heavily augmented aircraft. These latter points stem from the peculiarities of particular sensors or flight control system architectures. Finally, the consequences of increased time lags introduced by the control systems is addressed as a general issue associated with heavily augmented aircraft.

The final section (V) presents a summary of conclusions, and an appendix documents the generic RSS transport aircraft dynamics used in the study.

SECTION II

THE INCENTIVE FOR HEAVY AUGMENTATION

A. STABILITY AUGMENTATION

Since the immediate post-World War II period, stability augmentation has offered a standard means to improve the flying qualities of piloted aircraft. Augmentors modify the effective dynamics of the aircraft exhibited to the pilot. They operate in series with the pilot's inputs so that the pilot is unaware that an automatic system is involved. Thus, within the range of the augmentor's control authority, the effective aircraft dynamics can be adjusted over very wide ranges to best fit the pilot's abilities and desires.

Because of the series installation, safety in the event of augmentor failure has always been a central issue. With single-thread augmentors, restricted authority is the only answer. This authority must be sufficiently limited to permit safe recovery by the pilot in the event of hard-over failures, and the aircraft with the augmentor inoperative must be safely flyable. As a practical matter single-thread systems are usually confined to the improvement of aircraft rotory damping characteristics or small lift adjustments (e.g., improvement of the damping in the dutch roll mode, maneuver load control, blended DLC, etc). For most present-day transport aircraft these are the major uses of stability augmentation. In almost all cases, aircraft designers have attempted to make the aircraft safely flyable with stability augmentation off so as to avoid making the augmentor a dispatch item.

If a desire for other than simple damping functions or small lift changes arises, a larger proportion of total control surface authority is required for the augmentation. Historically, this has proceeded in several steps (Reference 1). Initially, the safety in the event of a hard-over was improved by using dual channels with the actuators summing their forces at a common point. In the event of a hard-over failure in

one channel, the other would then resist and counter, and the system would be "fail-soft" rather than hard-over. Thus, no untoward moment is applied to the aircraft, although the dynamics after failure are those of the aircraft alone. Then, in the most elaborate modern systems, such as those on the F-16, F-18, and Space Shuttle, the augmentor is multiply redundant, with either three or four largely independent channels. These systems are of very high integrity and are typically arranged to be multiple fail-operative. Essentially, any imaginable aircraft-alone stability deficiency can be corrected with such full-authority systems, as long as sufficient control power is available.

While such extensive multiple redundant, fly-by-wire (or fly-by-light) systems are currently part of many high-performance aircraft, they have yet to find a role in transports as full-time operating systems. Of course, fail-operational capability for relatively short-time tasks, such as autoland, is commonplace (e.g., the Trident has made more than 50,000 in-service automatic landings, see Reference 2). But, these systems are automatic flight controls (usually installed in parallel and moving the controls in lieu of the pilot) rather than augmentation systems, part-time rather than full-time, and always optional.

Although the stability problems of supersonic transports have led to considerations of multiple-redundant augmentation systems, such as the four independent channel hardened stability augmentation system (HSAS) considered for the US SST (References 3 and 4), for subsonic transports aircraft designers have been able to get by with nothing more elaborate than dual-channel dampers. In the main, these systems improve damping, fatigue life, ride qualities, and pilot workload. They have been non-flight-critical in that complete loss of function would result in increased crew workload but not in any hazard to continuous safe flight. And, most important to some of the considerations to be developed here, the effective dynamics of the augmented aircraft can be described as conventional for both attitude and path control functions. Consequently, for these types of aircraft, the FARs related to flying qualities are just as appropriate for stability augmented aircraft that behave conventionally as for conventional aircraft with no augmentation.

The primary additional considerations introduced by the presence of augmentation are associated with failure possibilities and recovery and continued safety of flight after failure. In the context of the FARs this basically means an extension of the number of flight tasks to be considered — tasks akin to those needed for proper assessment of sudden engine-out conditions in multi-engine aircraft.

B. AIRCRAFT CONFIGURATION TAILORING FOR ENERGY EFFICIENCY

The long-term status quo wherein stability augmentation has been desirable and cost-effective but nonetheless a non-flight-critical feature may be nearing an end. The progress of technology for multiple-redundant fail-operational systems and experience in many operational aircraft have been accompanied by major expansions in the activities which can be accomplished by flight control. These include a cornucopia of functions intended to permit extensions in performance envelopes — longitudinal and lateral stability enhancement, span load modification, elastic mode suppression, ride smoothing, flutter prevention, etc. These have been studied, and to some extent applied, over the last decade and are grouped under the general heading of active control technology (References 2, 5–9). Military and space applications have of course led the way, although the National Aeronautics and Space Administration and the transport aircraft manufacturers have devoted considerable effort to commercial transport applications, primarily in an attempt to reduce direct operating costs via fuel savings (e.g., References 10–12). This thrust toward energy-efficient operations has focused on drag reduction and L/D improvement. Some of the aircraft longitudinal configuration characteristics directly associated with stability and control which can effect these benefits are typified in Figure 1. Such features as reduced tail size can also have an impact on the lateral properties. The configuration characteristics listed have both aerodynamic and structural aspects. Parasite and trim drag reductions are predominantly aerodynamic, whereas L/D improvement and lift distribution adjustment involve structural features as well. For instance, the increased aspect ratio is achieved with significantly less

added structural weight than would normally be required by resorting to active control for wing load alleviation. Weight savings thus add to the aerodynamic improvement as an additional benefit.

Drag Reduction

Parasite drag -- reduced tail size

Trim drag -- reduced tail download and trimming surface(s) deflection

Induced drag -- more positive tail lift for given total trimmed lift coefficient

More elliptical lift distribution -- active control on wing

L/D Improvement

Increased aspect ratio -- active control on wing to reduce wing root bending moment

Figure 1. Some Aircraft Configuration Characteristics Associated with Improvements in Energy Efficiency

While these aircraft configuration characteristics provide fuel efficiency payoffs, there are control system "costs" incurred. The first is the need for a new control system to provide the active control features on the wing. The second is an extension in function and control authority of the longitudinal stability augmentor to redress the stability and control deficiencies caused by configuration tailoring for drag reduction and the active controls on the wing.

While all of the longitudinal stability characteristics are affected to some extent by the configuration characteristics, the major impact of

the Figure 1 trends is on the longitudinal weathercock tendency (short-period static stability) and the pitch damping. The principal aerodynamic stability derivatives governing these response properties are the pitching acceleration due to angle of attack, M_α (C_{m_α} in coefficient form) and the pitching acceleration due to rate of pitch, M_q (C_{m_q} as a coefficient). The pitch damping term, M_q, is reduced by the reduced tail size and by any reduction in effective tail length. This damping reduction could easily be countered by a restricted authority pitch damper. The more profound effect is on weathercock stability, M_α, which governs the static stability of the airplane. For trim drag reduction alone, the static stability would be made essentially neutral, but the wing load alleviation and other active control features can provide an additional destabilizing increment. Thus, the effective M_α can reflect from essentially neutral to unstable static margins. The rectification of these stability and control deficiencies and the provision of adequate flying qualities can no longer be accomplished by a limited authority damper, but now will require a much higher authority augmentation system. Thus, a multiple-redundant command augmentation system is indicated.

C. FAILURE LEVELS

The actual aircraft performance improvements available by application of integrated active controls including high-authority augmentation of the longitudinal characteristics are highly configuration-specific. Yet extensive studies have shown significant gains, especially with aircraft initially designed with all these features in mind. The studies to date (e.g., References 7, 9-13) indicate that those applications of active control technology that yield the greatest benefits will generally operate continuously and typically are essential in at least some flight regimes. For safety of flight considerations the criticality levels ("airplane functions" in FAA terminology) can be defined as indicated in Figure 2. Reliability objectives in terms of frequency of occurrence of

LEVEL	DEFINITION	RELIABILITY (FAILURES/ FLIGHT HOUR)
Non-essential	Functions which could not significantly degrade the capability of the airplane or the ability of the flight crew to cope with adverse operating conditions if accomplished improperly or lost. Failure conditions which result in improper accomplishment or loss of non-essential functions may be probable.	Probable $> 10^{-5}$
Essential	Functions which would reduce the capability of the airplane or the ability of the flight crew to cope with adverse operating conditions if accomplished improperly or lost. Failure conditions which result in improper accomplishment or loss of essential functions must be improbable.	Improbable $< 10^{-5}$ $> 10^{-9}$
Critical	Functions which would prevent the continued safe flight and landing of the airplane if not properly accomplished. Failure conditions which result in improper accomplishment or loss of critical functions must be extremely improbable.	Extremely Improbable $< 10^{-9}$

Figure 2. Flight Criticality Levels

failures per hour of flight time may be correlated as shown in Figure 2 through the requirements of Section 25.1309 of the FAR. Notice that the reliability for a "critical" system is tantamount to no failures within the lifetime of an aircraft fleet. There are no automatic flight control systems which have actually exhibited or have been realistically designed to this level of reliability, so augmentation systems which are critical may seem to be premature. On the other hand, systems which

operate at the essential level, where crew action can offset hazards imposed by failure, are completely feasible. As a practical matter, augmentors which are constrained to be no more than essential types will impose a limit on the degree of static instability permitted on the aircraft alone. That is, the pilot must be able to recover from a complete failure of the augmentation and retain safe control of the aircraft through its flight phases thereafter in spite of the instability.

In summary, the type of longitudinal stability augmentation needed to take full advantage of potential energy efficiency improvements will require large control authority and will operate full time. Hence it will be essential in nature. The aircraft without augmentation will in many flight conditions be unstable or nearly so. Accordingly, the stability augmentation system will be at least fail-operational for the first failure and may require an even higher degree of integrity if maximum performance benefits are to accrue from the configuration design. We refer to herein to the totality of these features as heavy augmentation to distinguish it from the more usual limited authority, dual or single thread lift incrementing or damper-type systems.

SECTION III

FLYING QUALITIES AND DYNAMICS OF RELAXED STATIC STABILITY AIRCRAFT

A. FLYING QUALITIES — GENERAL

"Flying qualities" are those properties of an aircraft which interact directly with the pilot's control actions in aircraft flight operations. As an object of control, the airplane dynamic characteristics can be divided into three general categories as a function of pilot control involvement (Figure 3). The first control function, unattended operation, depends predominantly on the stability and response to disturbances of the effective aircraft, i.e., the airplane dynamics as modified by the stability augmentation system. These properties are particularly important for ride qualities in permitting the crew to perform functions other than control. The second function, trim management, involves the pilot and effective airplane interactively, but only on a nearly static basis. Pilot involvement is highly intermittent and just sufficient to modify, adjust, and establish a condition of equilibrium flight. The effective aircraft characteristics of greatest importance in trim management are the steady-state or very-low-frequency dynamics of the aircraft plus augmentation system. Finally, in continuous operation, the pilot is participating as part of a closed-loop pilot/vehicle system in changing the aircraft flight path (maneuvering) or maintaining a desired flight path in the presence of disturbances (regulation). The effective aircraft dynamics involved in continuous control are all of the rigid-body modes of the heavily augmented aircraft, as well as some higher-frequency filtering, delay, and flexible mode dynamics. In other words, the closed-loop pilot/vehicle system control functions exercise all of the heavily augmented aircraft dynamics present within the total bandwidth over which the pilot can exert control.

PILOT INVOLVEMENT	CONTROL FUNCTION
None	Unattended operation
Intermittent	Trim management
Continuous	Maneuvering Regulation

Figure 3. Control Functions in Flight Operations

In the context of the FARs the control functions of Figure 3 are exercised under any probable set of operating conditions and aircraft configurations (including failures and emergencies) through the five flight phases of:

- Takeoff
- Climb
- Level flight
- Dive/descent
- Landing/go around

The FARs in essence require that the control functions be accomplished without danger in all these phases and without requiring "exceptional piloting skill, alertness, or strength." Although the FARs are largely qualitative, in the final analysis there is a basic "quantification" of the aircraft flying qualities delivered by the FAA test pilot cadre as a go/no-go assessment of the level of skill, attention (alertness), and strength required in performing required tasks. For heavily augmented aircraft with fully operative augmentation, such assessments may be the same in principle to those conducted with conventional aircraft. However, there can be, as we will develop below, sets of effective aircraft dynamics for heavily augmented vehicles which differ in kind and degree from those of conventional aircraft. These differences may significantly affect the pilot's assessment. A major purpose of this report is to expose and describe these differences.

B. EXPANSION OF OPERATING CONDITIONS/ CONFIGURATIONS DUE TO SAS FAILURE STATE

The presence of heavy augmentation can have another major impact on flying qualities when all probable operating conditions and configurations are taken into account. The stability augmentation of a heavily augmented aircraft will often be operating at the essential level wherein complete loss of the augmentation function can result in a potential hazard to safe flight which can be averted by appropriate flight crew action. In assessing the flight safety of such systems, the spectrum of SAS failure possibilities must be carefully delineated and examined for all flight phases. From such an examination the most critical condition(s) can then be defined as test configurations just as engine-out conditions are currently defined.

The myriad failure possibilities of multiple-redundant stability augmentation systems make the determination of failure possibilities and probabilities an extremely tedious and demanding analysis task. Even with great skill and persistence, some possibilities seem inevitably to be overlooked, yet such failure mode and effects analysis and reliability estimates must be accomplished to establish the general status of affairs with an essential or critical system. From the flight safety standpoint, the key is to determine those effective vehicle dynamics which are most critical from the standpoint of potential hazard to continued safe flight and required crew action to avert the hazard. As a practical matter, this failure criticality assessment should be considered almost independently from its probability of occurrence (although clearly complete loss of function in an essential system should be remote). If safe controllability and maneuverability can be demonstrated in the presence of the most difficult control hazard(s) without requiring exceptional piloting skill, alertness, or strength, and without danger of exceeding the limit load factor, then surely the aircraft with failed essential SAS would meet the intent of the FARs. By focusing upon the conditions of augmentation failure which result in the worst conceivable control tasks all manner of sophistic discussion about reliabilities and probabilities may be short-circuited, if not avoided completely.

This approach logically applies at the essential level, where demonstration that crew action can avert any conceivable hazard follows from the flight-critical definition. If the SAS is critical in character, such that complete loss of the SAS functions, however improbable, results in immediate and unconditional hazard to continuous safe flight, the considerations become far more involved. In principle, there are then no tasks analogous to engine failure which can be specified for consideration in the event of complete loss of function. Again, there are myriad conditions of partial function loss for which the effective airplane dynamics may be considered as appropriate tasks for pilot assessment, and for which the approach indicated above for essential portions applies. But complete loss of augmentation function is by definition intolerable and the burden of proof requires a different fundamental approach from that currently embedded in the flying qualities portions of the FARs. In other words, the current ultimate appeal to flight demonstration inherent in the FARs cannot resolve all the issues with critical systems. Fortunately, for subsonic Part 25 aircraft of the future, only essential functional levels are likely (although the systems may have reliability tantamount to critical levels).

C. ALLOWABLE SAS-OFF AIRCRAFT DYNAMICS

The actual determination of appropriate tasks to assess for safe recovery and flight in the event of various essential system failures is system-specific, so we will not treat it further here. However, the controllability of near-neutral and unstable aircraft, which may be representative of the post-failure-of-function condition when the effective dynamics of the heavily augmented aircraft become those of the aircraft alone have been studied extensively for well over two decades (e.g., References 14-18). These investigations apply primarily to fail-soft situations wherein the loss of SAS function results only in a change in the aircraft dynamics without the application of an additional transient failure moment as well. The effect of "surprise" failures such as sudden loss of SAS function was not thoroughly evaluated in any of the studies.

The latest review of all available data (Reference 18) fails to provide an unequivocal quantitative basis for a standard. The data do indicate that some slight short-period divergence can be safely countered by the pilot; that workload is significantly increased as the divergence time decreases; and that there appears to be little difference in pilot assessment of workload between normal and emergency operations. In the case of the latest proposed military specification (Reference 17), when the statement of a quantitative criterion was deemed essential, a divergence time to double amplitude of 6 seconds for Level 3* conditions was selected (supported again by the data alluded to above).

A rigid constructionist reading of the FARs indicates no longitudinal divergence is permissible. On the other hand, the US SST (e.g., References 3 and 4) and the Concorde (e.g., Reference 19) either planned to permit or actually exhibit a divergence with about a 6 second time to double amplitude.

The above status indicates that the FARs as interpreted for the failure states of essential systems can probably be relaxed in the sense that a slight instability after failure can be tolerated. However, more data specifically oriented to the crew functions and behavior in takeover, recovery, and continued safe flight with unexpected SAS failures are needed before a definite quantitative statement (in terms of such parameters as time to double amplitude) can be made.

D. HIGH WORKLOAD PILOT/AIRCRAFT CLOSED-LOOP CONTROL OPERATIONS

To assure safety, those aircraft control functions which demand the greatest pilot attention and skill require primary consideration in

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*Level 3 is defined as "flying qualities such that the airplane may be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both....Category B and C Flight Phases [which include Climb, Cruise, and Descent and Takeoff, Approach, Go-Around, and Landing] can be completed."

flying qualities assessments. These are important not only because the piloting demands must be consistent with flight operations which do not require "exceptional piloting skill, alertness, or strength," but also because the high control workload situations reduce the pilot attention available for other activities. This leads to a most important indirect assessment that high control-associated workload is not detrimental to the pilot's ability to perceive and cope with the unexpected.

The most demanding high workload pilot/aircraft closed-loop control operations tend to be those which involve precision path control in unfavorable environmental conditions. All flight phases involve some form of path control which incorporates both flight path changes and flight path maintenance or regulation. In most ordinary flight circumstances path control, while an essential pilot skill, is nonetheless a relatively benign and low-stress function. On the other hand, when the path task is itself very demanding and the environment unfavorable (e.g., low visibility approach and landing in turbulence and shear), the precision path control task becomes exceedingly exacting. For these reasons we shall here use precision path control as the control task to explore the effects of heavy augmentation on closed-loop pilot/aircraft system flying qualities.

A block diagram that indicates the pilot's activities in precision path control is shown in Figure 4. On the right the augmented aircraft has path deviation and pitch attitude as the output variables stemming from aircraft dynamics which are forced by external atmospheric disturbances, η, and the pilot control output, δ. The augmented aircraft itself is a closed-loop system comprising the airplane-alone and augmentation system. Thus, the sensors, computation and actuation elements involved in the feedback control augmentation system, as well as the aircraft alone, are encompassed by the "augmented aircraft" single block in Figure 4. An underlying assumption in this diagram is that other aircraft control effectors such as throttle or flap are not being continuously modulated by pilot control action. (Trim management using these aircraft effectors, however, is not excluded.)

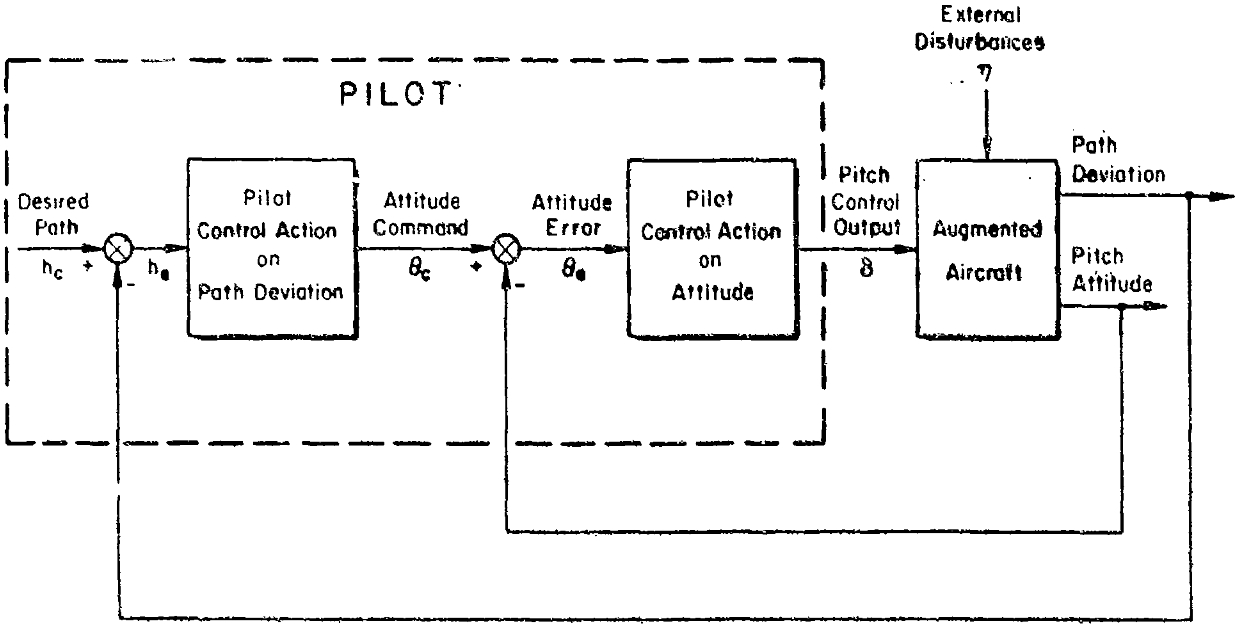


Figure 4. Closed-Loop Pilot-Aircraft System for Precision Path Control

Even though trimmed precisely, the augmented aircraft will not by itself maintain exactly the prescribed path and attitude in the presence of disturbances. Consequently, the pilot must exert control not only to establish the desired path but also to correct any deviations from the desired attitude and path. This is accomplished by the pilot acting as a closed-loop controller, which means simply that the pilot's control output is dependent on (i.e., a function of) path deviation and attitude. Thus, a component of the pilot's control output is correlated with an attitude error and another component is correlated with the difference between the desired and actual path. This relationship is depicted in the Figure 4 block diagram as a so-called "series" pilot closure, i.e., the pilot's action on path deviation acts in series with, and provides an internal "attitude command" for, the pilot's action on attitude error. Several research studies using elaborate and detailed measurements of just this situation (e.g., References 20 and 21) indicate that this series structure and general pilot behavior control model is appropriate for path control situations. In essence, the pilot's

higher-frequency control actions are devoted primarily to attitude so that the "inner" attitude loop is tightly closed and the attitude is well regulated. This tight inner loop makes possible the effective closure of the "outer" path deviation loop without excessive pilot equalization or compensation. Without good control of attitude the pilot would have to be way ahead of any path changing trends, requiring very difficult anticipation and high workload. (Examples include altitude control using only airspeed and altimeter or control during approach using only airspeed and the raw ILS glidepath data.) If the attitude loop is difficult for the pilot to interact with and close (i.e., if the augmented aircraft pitch attitude dynamics are deficient in that they require excessive pilot compensation and attentional workload), attitude control will suffer directly and path control indirectly.

To focus more on the aircraft characteristics, the Figure 4 block diagram is expanded somewhat in Figure 5. Here the pilot's activities

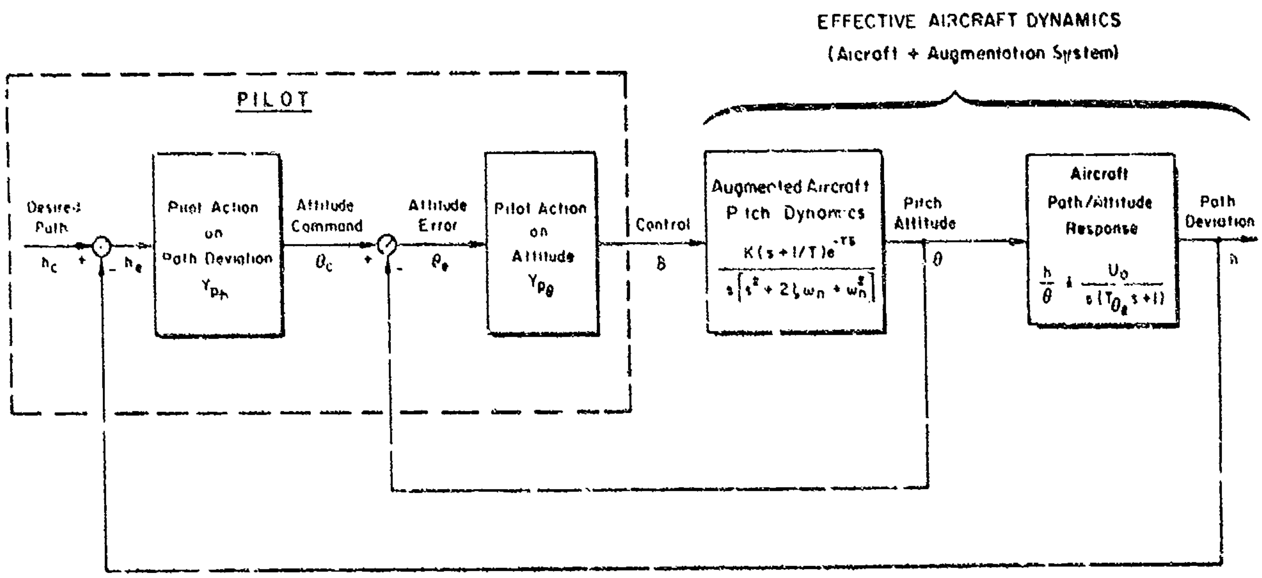


Figure 5. Closed-Loop Precision Path Control with Attitude Control Inner Loop

are shown as before, with the addition of symbolic transfer characteristics Y_{p_h} and Y_{p_θ}. These characterize quantitatively the pilot's input/output relationships between path deviation error h_e and attitude command θ_c, and attitude error θ_e and pilot control output δ. In any detailed quantitative analysis of the actual closed-loop pilot/vehicle system these transfer characteristics can be used to describe and quantify the pilot's activity. The primary difference between Figs. 4 and 5 is in the augmented aircraft dynamics, which are broken down into two components in Fig. 5. The first block, the augmented aircraft pitch dynamics, relates pitch attitude, θ, to pilot control, δ. In the absence of any other inputs to the aircraft, the pitch attitude and path deviation are dynamically related by a direct input/output entity shown in the Path/Attitude Response block.

The aircraft blocks show the transfer characteristics in fairly general form for the pitch attitude dynamics and for the path/attitude response dynamics. These will be discussed in depth below. It will evolve, in fact, that a key issue in the differences between the flying qualities of conventionally augmented aircraft and very heavily augmented aircraft resides in the <u>differences</u> in the transfer characteristics for the augmented pitch attitude and the <u>similarities</u> for the path/attitude response. By focusing on these facets we can expose the major differences between heavy and conventional augmentation without an elaborate argument involving the pilot's detailed control actions. It is extremely important to recognize, however, that the closed-loop aspects are a central issue in that the pilot's assessment of the suitability of the aircraft inherently depends upon his actions required to accomplish control. (As an adaptive controller the pilot adjusts his control actions as needed to compensate for the aircraft dynamics; so different aircraft dynamics mean different pilot actions and different pilot assessments.) Thus, the feedback loop structure and what the pilot actions are on path deviation and attitude are of central concern to set the context of the control task. Yet, within this context, we can focus primarily on the augmented aircraft pitch dynamics and the aircraft path/attitude response to explore differences between conventional and heavy augmentation.

B. CONVENTIONAL AIRPLANE-ALONE DYNAMIC CHARACTERISTICS

The constant speed, so-called short-period dynamics of the aircraft provide an adequate structure on which to build our discussion of aircraft-alone dynamics, including conventional augmentation effects. Two degrees of freedom of aircraft motion are involved — vertical velocity, w (or "inertial" angle of attack, $\alpha = w/U_0$), and pitch attitude, θ. The equations relating these variables with the control δ are derived and discussed in detail in texts on aircraft dynamics (e.g., Reference 22). Typical results are summarized in Figure 6a. In the short-period equations the aerodynamic characteristics in the vertical acceleration equation include the accelerations $Z_w w$ and $Z_\delta \delta$. These characterize heave damping, Z_w, or alternatively lift changes due to changes in angle of attack ($Z_\alpha = U_0 Z_w$) and the vertical acceleration (lift per unit mass) due to control deflection, $Z_\delta \delta$. The aerodynamic terms in the pitching acceleration equation are the pitch damping, M_q, and weathercock stability effect, M_w ($M_\alpha = U_0 M_w$), previously referred to as major factors in relaxed stability airplane dynamics. s is the Laplace transform variable and can be taken to be the equivalent of the derivative operator d/dt. The $sw + U_0 s\theta$ components of the vertical acceleration equation are an inertial and centripetal acceleration, respectively. Similarly, the $s^2\theta$ (equivalent to $d^2\theta/dt^2$) represents the inertial pitching acceleration. $M_{\dot{w}} sw$ is a relatively small aerodynamic pitching acceleration component proportional to the rate of change of angle of attack. An auxiliary equation relating the kinematics, such as normal acceleration a_z, and flight path angle, γ, is also given in Figure 6a.

The equations can be converted to transfer functions by solving for θ, h, γ, etc., as a function of the control input δ. The transfer function relating pitch rate to control deflection is shown in Figure 6b, where the alternative symbol q for pitching velocity is introduced for $s\theta$. The transfer function relating altitude to control deflection is also given in Figure 6b. It will be noted that the second (approximate) relationship listed does not include the higher-frequency terms (those involving s^2 and s) in the numerator. These typically are important at frequencies beyond the bandwidth of pilot control interest, although

(a) SHORT PERIOD EQUATIONS

Vertical Acceleration: $(s - Z_w)w \quad - \quad U_o s\theta \quad = \quad Z_\delta \delta$

Pitching Acceleration: $-(M_{\dot{w}}s + M_w)w + (s - M_q)s\theta \quad = \quad M_\delta \delta$

Kinematic Relationship: $a_z \quad = \quad - s^2 h \quad = \quad sw - U_o s\theta \quad = \quad U_o s\gamma$

(b) TRANSFER FUNCTIONS

Pitch Attitude:

$$\frac{s\theta}{\delta} = \frac{q}{\delta} \doteq \frac{M_\delta\left[s + \left(-Z_w + \frac{Z_\delta}{M_\delta} M_w\right)\right]}{s^2 - \left(Z_w + M_{\dot{\alpha}} + M_q\right)s + \left(Z_w M_q - M_\alpha\right)}$$

$$= \frac{M_\delta\left(s + 1/T_{\theta_2}\right)}{\left[s^2 + 2(\zeta\omega)_{sp}s + \omega_{sp}^2\right]}$$

Altitude:

$$\frac{h}{\delta} = \frac{-Z_\delta\left[s^2 - \left(M_q + M_{\dot{\alpha}}\right)s - \left(M_\alpha - \frac{M_\delta}{Z_\delta} Z_\alpha\right)\right]}{s^2\left[s^2 - \left(Z_w + M_{\dot{\alpha}} + M_q\right)s + \left(Z_w M_q - M_\alpha\right)\right]}$$

$$\doteq \frac{-U_o M_\delta\left(1/T_{\theta_2}\right)}{s^2\left[s^2 + 2(\zeta\omega)_{sp}s + \omega_{sp}^2\right]}$$

Path/Attitude:

$$\frac{h}{\theta} \doteq \frac{U_o}{s\left(T_{\theta_2}s + 1\right)}$$

(c) DYNAMIC PARAMETERS

$$1/T_{\theta_2} = -Z_w + \frac{Z_\delta}{M_\delta} M_w \doteq -Z_w = \frac{\rho S U}{2m}\left(C_{L_\alpha} + C_D\right)$$

$$2(\zeta_\omega)_{sp} = -\left(Z_w + M_{\dot{\alpha}} + M_q\right)$$

$$\omega_{sp} = \sqrt{Z_w M_q - M_\alpha}$$

Figure 6. Constant Speed Airplane-Alone (Short Period) Relationships

they can be of marginal importance on, for instance, elevon controlled aircraft.

The last relationship of Figure 6b is the path/attitude response. It is seen from the transfer functions that their denominators are similar. When attention is focused on pilot control inputs, the path deviation h can be considered to be derived from pitch attitude on a cause/effect basis. Here pitch attitude is an input operating on a transfer function which is just the ratio $(h/\delta)/(\theta/\delta)$. The result is given in Figure 6b. The path deviation h obtained using this ratio will be valid for pitch inputs δ; an additional term must be added if any disturbance is present.

The time courses of the dynamic response of the output variables pitch attitude and path for a control input in constant-speed flight depend only on the three quantities T_{θ_2}, ζ_{sp}, and ω_{sp}, listed in Figure 6c. Speed, U_0, and control effectiveness, M_δ, operate as scale factors. ζ_{sp} and ω_{sp} are the short-period damping ratio and undamped natural frequency, respectively. The short-period frequency is governed primarily by the weathercock or static stability term, M_α. As expected, the pitch damping, M_q, is prominent in the short-period damping, $(\zeta\omega)_{sp}$. This is easily augmented artificially by feeding back a signal to the elevator which is proportional to q in the frequency region about ω_{sp}.

The time constant T_{θ_2} is sometimes referred to as the pitch attitude lead time constant because it appears in the θ/δ transfer function numerator and is hence a "lead" term. It is also often referred to as the path/attitude "lag" time constant. This stems from its appearance in the denominator of h/θ. If we imagine a step pitch attitude change, then the rate of change of path deviation, $\dot{h}$, or of flight path angle, γ, will respond exponentially, with the time lag T_{θ_2} (see Figure 7). Because $1/T_{\theta_2} \doteq -Z_w$, which in turn is proportional to the lift/curve slope of the airplane, C_{L_α}, this path-to-attitude time lag is a direct function of the fundamental performance characteristics of the airplane. Once the wing is designed (and C_{L_α} set), the flight path lag, T_{θ_2}, for a given flight condition cannot be changed without appealing to direct-lift control of some sort. As a flight path lag T_{θ_2} is an extremely

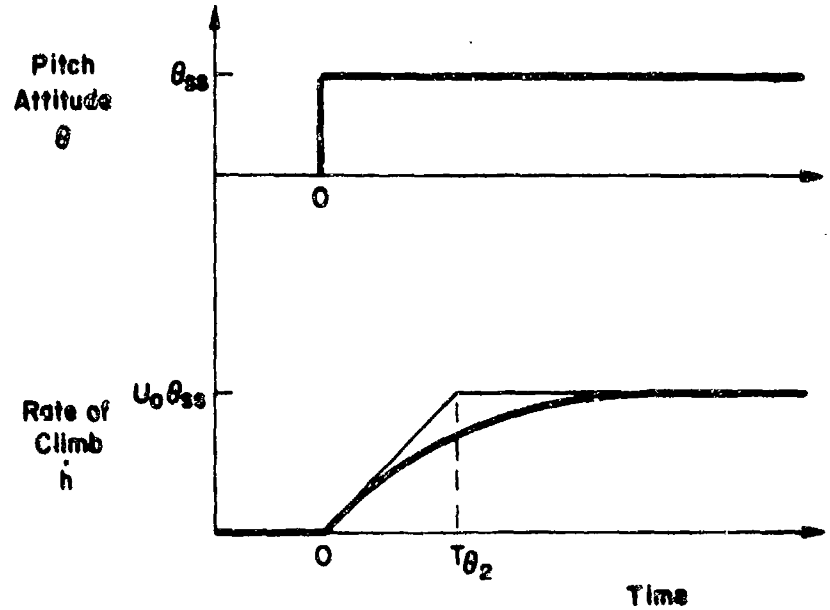
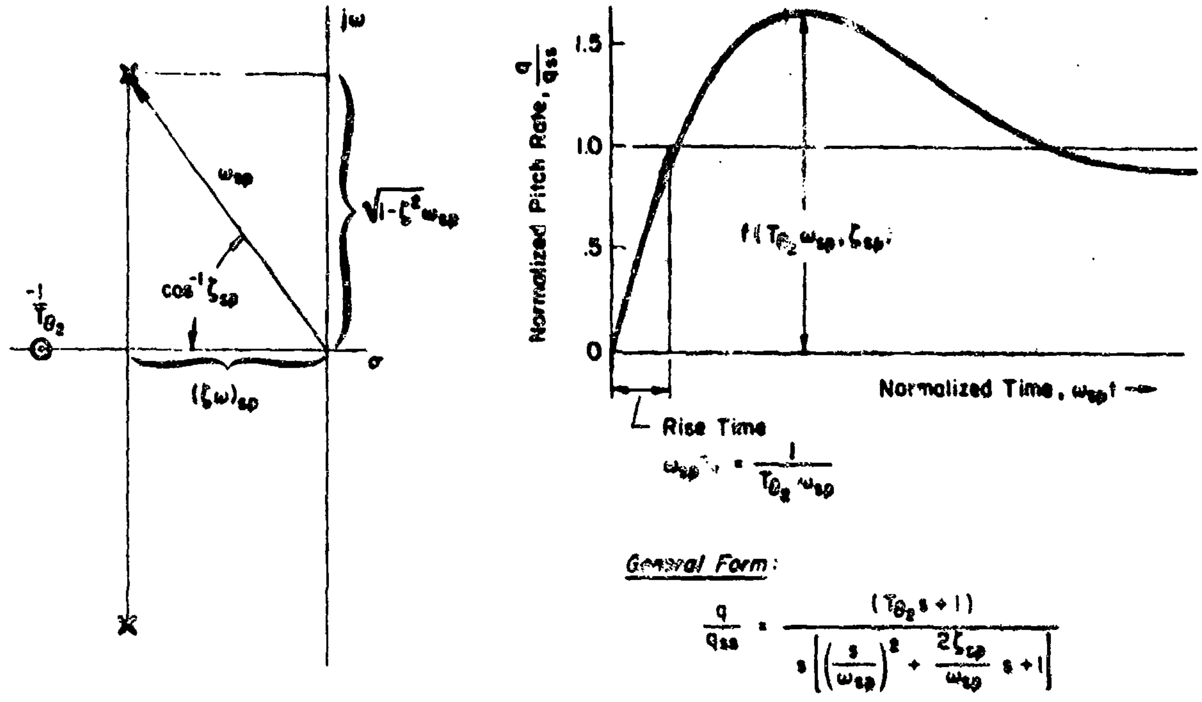


Figure 7. Path Rate Response to Attitude Change

important factor in the attainable path precision as it limits the pilot's control bandwidth even if the pitch attitude dynamics are perfect. Thus, a large T_{θ_2} will inevitably result in a path response which is more sluggish than that for a configuration wherein T_{θ_2} is a good deal smaller.

F. SHORT-PERIOD ATTITUDE DYNAMICS FOR RELAXED STATIC STABILITY AIRCRAFT

For a conventional subsonic aircraft the center of gravity is ordinarily located well ahead of the neutral point (that c.g. location for which $M_\alpha = 0$ and the aircraft is neutrally stable or neutrally statically balanced). The pitching moment due to angle of attack is then highly negative, and the undamped natural frequency, ω_{sp}, may be fairly large. As numerical examples, for transports Reference 17 requires $\omega_{sp} > 1$ rad/sec for up-and-away flight conditions and $\omega_{sp} > 0.7$ rad/sec for approach. The short-period damping ratio is typically about 0.5 or somewhat higher when a pitch damper is installed. For these conditions a root plot which shows the location of the poles and zeros of the q/δ transfer function might appear as in Figure 8a. The pitch attitude zero



a) s-Plane Root Plot for q/δ

b) Pitch Rate Response to Step Control Surface Input

Figure 8. Short-Period Attitude Dynamics for an Aircraft with Large Static Stability

at $-1/T_{\theta_2}$ is shown by the circle, whereas the short-period poles are given crosses. Since the poles occur as a conjugate complex pair, the upper half of the s-plane plot and the lower half are the same. (In subsequent root plots only the upper half will be shown.) The undamped natural frequency, ω_{sp}, is the length of the vector from the origin of the s-plane plot to the location of a quadratic pole. This vector makes an angle with the real axis given by $\cos^{-1}\zeta_{sp}$. The short-period roots may also be thought of in terms of the damping, $(\zeta\omega)_{sp}$, and damped natural frequency, $\omega_{sp}\sqrt{1-\zeta_{sp}^2}$, which comprise their rectangular coordinates.

The aircraft transient response characteristics which correspond to this s-plane plot might appear as shown in Figure 8b. On this figure the pitching velocity, q, for a step elevator is given in normalized

form, scaled by the steady-state pitching velocity, q_{ss}. The time is non-dimensionalized using $\omega_{sp}t$ as the time variable. Notice that the non-dimensionalized rise time, $\omega_{sp}T_r$, depends only on the product of the lead time constant, T_{θ_2}, and the short-period undamped natural frequency. Interestingly enough, the peak overshoot also depends only on this product and the damping ratio, ζ_{sp}. Figure 9 (taken from Reference 23) shows this overshoot as a function of the $T_{\theta_2}\omega_{sp}$ product, with ζ_{sp} as a parameter. Because of the $(T_{\theta_2}s + 1)$ lead in q/q_{ss}, there is an overshoot even when the damping ratio is critical ($\zeta_{sp} = 1$). For instance, an overshoot of 2 occurs for $\zeta_{sp} = 1$ when $1/T_{\theta_2}\omega_{sp} \doteq 0.23$. For damping ratios less than 1 the overshoot of course becomes larger still, approaching nearly 5 for this instance as ζ_{sp} approaches zero. The important physical point to be gained from Figure 9 is that, for nominal well-damped cases, say $\zeta_{sp} > 0.5$ or so, the overshoot depends primarily on the separation between $1/T_{\theta_2}$ and ω_{sp}. $1/T_{\theta_2}$, being dependent on the lift curve slope, C_{L_α}, as scaled by speed and density, is an aircraft variable set mainly by wing configuration, whereas ω_{sp} can be adjusted by shifting the aircraft's balance, i.e., the c.g. location. The more forward the c.g., the greater the weathercock stability, and the stiffer the short period mode; hence, from Figure 9, the greater the overshoot (for a given $1/T_{\theta_2}$ and ζ_{sp}). Similarly, the rise time decreases as the short-period undamped natural frequency is increased. Rise time is given approximately by:

$$T_r \doteq \frac{1}{T_{\theta_2}\omega_{sp}^2}$$

$$\doteq \frac{Z_w}{M_\alpha}$$

Lift curve slope and rise time thus vary together, as does the peak overshoot for a fixed ω_{sp}.

As an aside before proceeding further let us put these results in context with the closed-loop precision path control of Figure 5. The aircraft path/attitude response and the maximum attainable tightness of

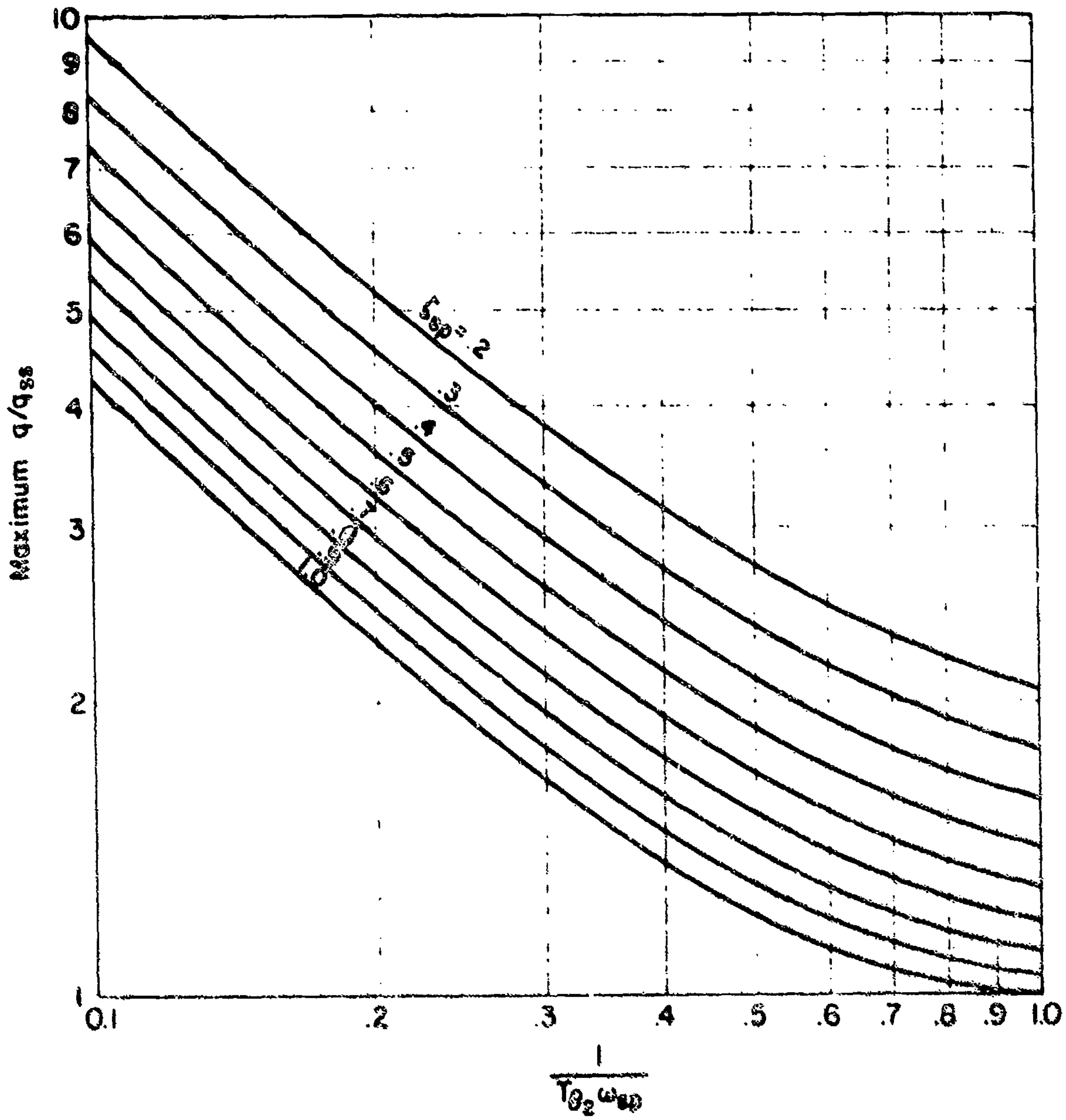


Figure 9. Maximum Pitch Rate Overshoot for Step Control Input

control of the outer path deviation loop are, as already remarked, limited by the path/attitude lag, T_{θ_2}. For conventional airplanes, the augmented aircraft pitch dynamics lead has the same time constant, T_{θ_2}. (In other words, the $1/T$ in the augmented aircraft pitch dynamics in Figure 5 is $1/T_{\theta_2}$.) Thus the attitude lead and the path lag are the same quantity, fixed by the same aircraft configuration feature (the lift curve slope). The undamped natural frequency and damping ratio of the augmented aircraft pitch dynamics are, in this case, $\zeta = \zeta_{sp}$ and $\omega_n = \omega_{sp}$. Consequently, for this conventional aircraft the pitch and path dynamics are predominantly dependent on these three variables, T_{θ_2}, ζ_{sp}, and ω_{sp}, which in turn depend on the lift curve slope, the weathercock stability, and the pitch damping.

The variation of the aircraft short-period characteristics as static stability is reduced can easily be studied using a root locus approach. The idea is to examine root plots, such as Figure 8a, as M_α is varied. This is done by plotting the short-period roots for a number of given values of M_α, and then connecting the roots to form a locus.

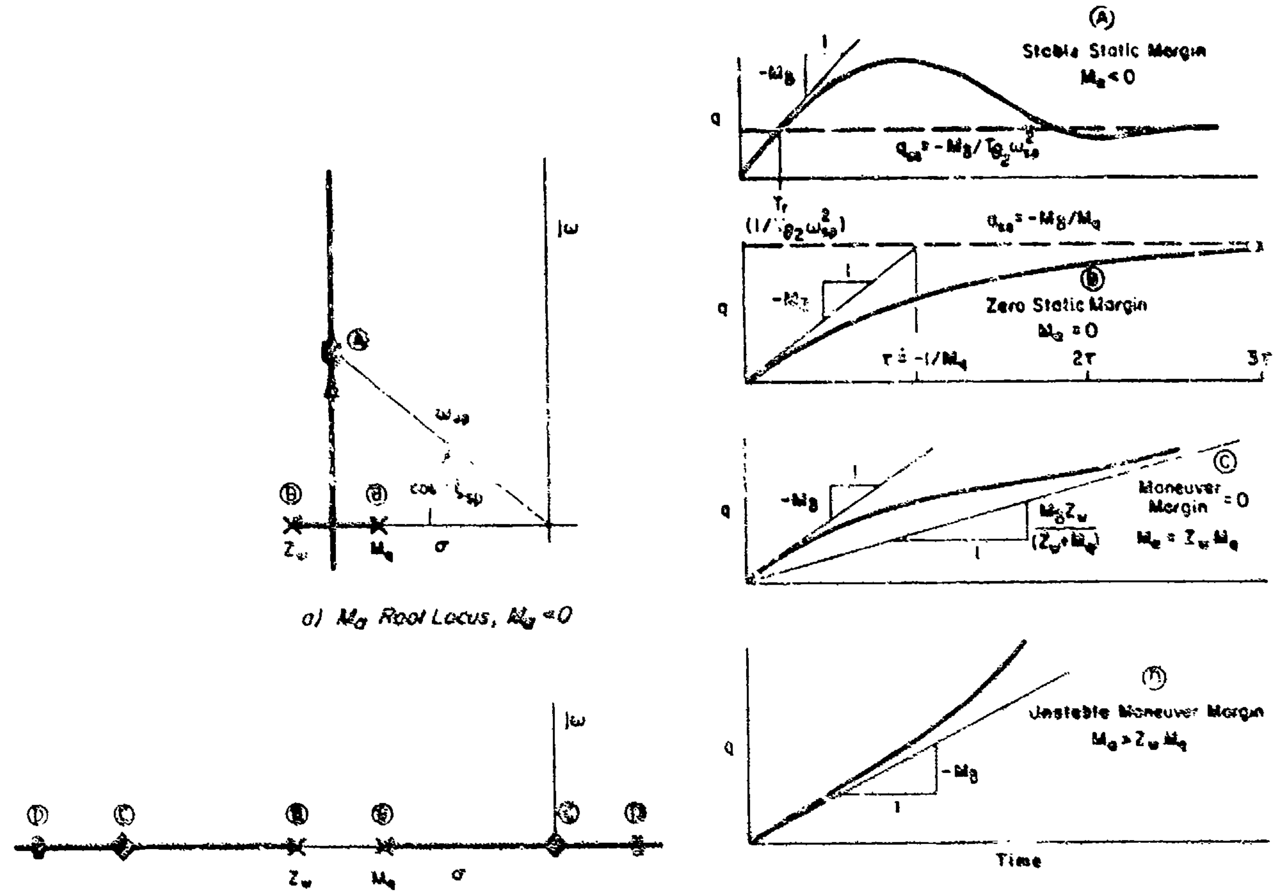
The short-period characteristic function is given approximately by:

$$s^2 + 2(\zeta\omega)_{sp}s + \omega_{sp}^2 = s^2 - \left(Z_w + M_{\dot{\alpha}} + M_q\right)s + \left(Z_w M_q - M_\alpha\right)$$

$$\doteq \left(s - Z_w\right)\left(s - M_q\right) - M_\alpha$$

When the static stability is zero, i.e., the c.g. is at the neutral point and $M_\alpha = 0$, the short-period characteristic function reduces to $(s - Z_w)(s - M_q)$. These roots are used as starting points for the locus. The effects of M_α variation on the short period are shown as root and corresponding step function control input time response plots in Figure 10.

Figure 10a illustrates the root variation as M_α is permitted to become more negative (e.g., c.g. increasingly moved further forward of the neutral point). The short-period changes from the two roots (Z_w and M_q) at B to a rendezvous point at $(Z_w + M_q)/2$, and then break away to



a) M_α Root Locus, $M_\alpha < 0$

b) M_α Root Locus, $M_\alpha > 0$

c) Pitch Rate Response To Step Elevator Inputs

Figure 10. Effect of Changes in M_α on Conventional Aircraft Short-Period Mode Characteristics

form an underdamped pair represented by the point at A. This corresponds to the normal root plot for an aircraft with large static stability, already discussed in connection with Figure 8. The time response sketch of A is qualitatively the same as that of Figure 8b, although the time history here is not normalized. The initial slope of response is given by the control surface effectiveness, M_δ.

For zero static margin ($M_\alpha = 0$), however, the short-period roots and the pitch rate response to step inputs are markedly different from highly stable conditions. The response is still stable, and the initial acceleration is still governed entirely by M_δ, but the shape is that

of a first-order exponential akin to that shown in Figure 10c for the path lag. The reason here is that the pitch attitude lead represented by the $-1/T_{\theta_2}$ zero almost exactly cancels the pole associated with $s - Z_w$. From Figure 6b the transfer function of q/δ is:

$$\frac{q}{\delta} \doteq \frac{M_\delta\{s + [-Z_w + (Z_\delta/M_\delta)M_w]\}}{(s - Z_w)(s - M_q)}$$

$$\doteq \frac{M_\delta}{(s - M_q)}$$

The time constant of this response, $-1/M_q$, is inversely proportional to the pitch damping term.

The nearly first-order type of response, shown as B in Figure 10c, is typical of the aircraft-alone responses for moderately relaxed static stability aircraft. Not only is the overshoot not present, but the time constant of the response may be quite large, especially when M_q is reduced over that of a normal airplane by cutting down on tail size. Even a conventional aircraft, operated with the c.g. near its neutral point, will appear more sluggish to the pilot than the same airplane operated with a highly stable static margin. Both the first-order form of the response, which exhibits no overshoot, and the value of $-1/M_q$ which is large when contrasted with typical spritely response times for the A type response, are associated with such assessments. The aircraft control sensitivity, indicated here predominantly by the $K_{ss} = M_\delta/M_q$, can also be markedly different from that for the nominal, highly stable response.

As the static margin is further reduced from zero by making M_α positive, the root locus shifts to Figure 10b. All the roots are now real. Static margin is proportional to M_α, whereas the maneuver margin is proportional to $Z_w M_q - M_\alpha$. $Z_w M_q$ is always positive, so the maneuver point, or c.g. location where the short-period undamped frequency becomes zero, is aft of the neutral point. The case for zero maneuver margin is shown by the root locations C and the time response C. The short-period roots in this instance are given by $s[s - (Z_w + M_q)] = 0$. The free s in this

characteristic function results in a ramp steady-state pitch rate for a step elevator, which has the slope per unit of elevator deflection shown in the time response. The initial slope is still M_δ, as it is for all cases. The middle portion of the response for zero maneuver margin looks very similar to that for zero static margin, in that it appears nearly exponential before proceeding toward its long-term ramp asymptote.

Finally, the aircraft c.g. may be in an unstable position as shown by the roots at D. One root is in the right-half plane, indicating an unstable divergence; the other one is somewhat larger in magnitude than $Z_w + M_q$. The initial transient associated with this root is therefore quite rapid, and the longer-term response is dominated by the unstable divergence departing more and more from the ramp defined by the initial pitch acceleration.

In order to achieve the maximum benefits of improved energy efficiency, the configuration changes listed in Figure 1 are appropriate. These in their totality lead to aircraft dynamics best represented by the pitch rate responses by B through D. As noted, these dynamic characteristics may be sluggish and insensitive at best, unstable and constantly demanding attention at worst. Pilot workload will be high in closed-loop control, while unattended operation may be a misnomer in that the pilot may always have to be attentive and alert to divergence tendencies. Because the RSS dynamics are not in general favorable, they will require correction using heavy augmentation.

C. SOME SPECIFIC EXAMPLES OF RELAXED STATIC STABILITY AIRCRAFT DYNAMICS

Most of the discussion to this point has emphasized a simplified approach to focus on key points and issues. This same viewpoint will be contained in subsequent articles, since many of these key points and issues have yet to be developed. At this juncture, however, it is pertinent to show some concrete results from a specific example of a relaxed static stability aircraft. For the purpose of this study a model of a hypothetical, high subsonic jet transport with relaxed static

stability was generated to be used as a basis for the exploration of potential flying qualities problems. This generic RSS transport was based in part on two designs from NASA Energy Efficient Transport studies — the Lockheed L-1011RE and the Boeing IAAC. The generic transport data base is described in Appendix A.

In the description of short-period dynamics we have emphasized the pitching velocity response. The angle of attack and flight path are also major response variables. The generic aircraft short-period characteristics have been used to develop some illustrative time histories for all the short-period variables. These are shown in Figure 11 for a typical cruise condition at M = 0.74 and 38,000 ft. Three static margins are considered: a stable margin where the c.g. is 7-1/2 percent of the mean aerodynamic chord, $\bar{c}$, ahead of the neutral point; zero maneuver margin, which corresponds to $-1.52\%\ \bar{c}$; and a $-5\%\ \bar{c}$ unstable static margin. The input is a horizontal tail (1 deg-sec) unit impulse. For such an input the pitch <u>attitude</u> responses in Figure 11 appear with the same response shapes as those described previously in connection with Figure 10 for pitch <u>rate</u> in response to a step elevator input.

For the stable aircraft, the pitch attitude response has an overshoot, as expected, due to the numerator lead, T_{θ_2}. Flight path angle is basically the pitch attitude response delayed by a lag, given by $1/(T_{\theta_2}s + 1)$. Because this lag cancels the attitude lead, the γ response builds up as a unit numerator second-order response approaching an equilibrium condition wherein $\theta = \gamma$. The pitch and flight path angle changes are accomplished aerodynamically by an angle of attack buildup and subsequent return towards zero.

The zero maneuver margin case conveniently divides the unstable and statically stable situations. Both pitch attitude and flight path angle approach constant rates of change. The angle of attack, on the other hand, exponentially approaches a constant value.

Finally, the 5 percent negative static margin responses for all three variables depart exponentially. Notice, however, that all of the

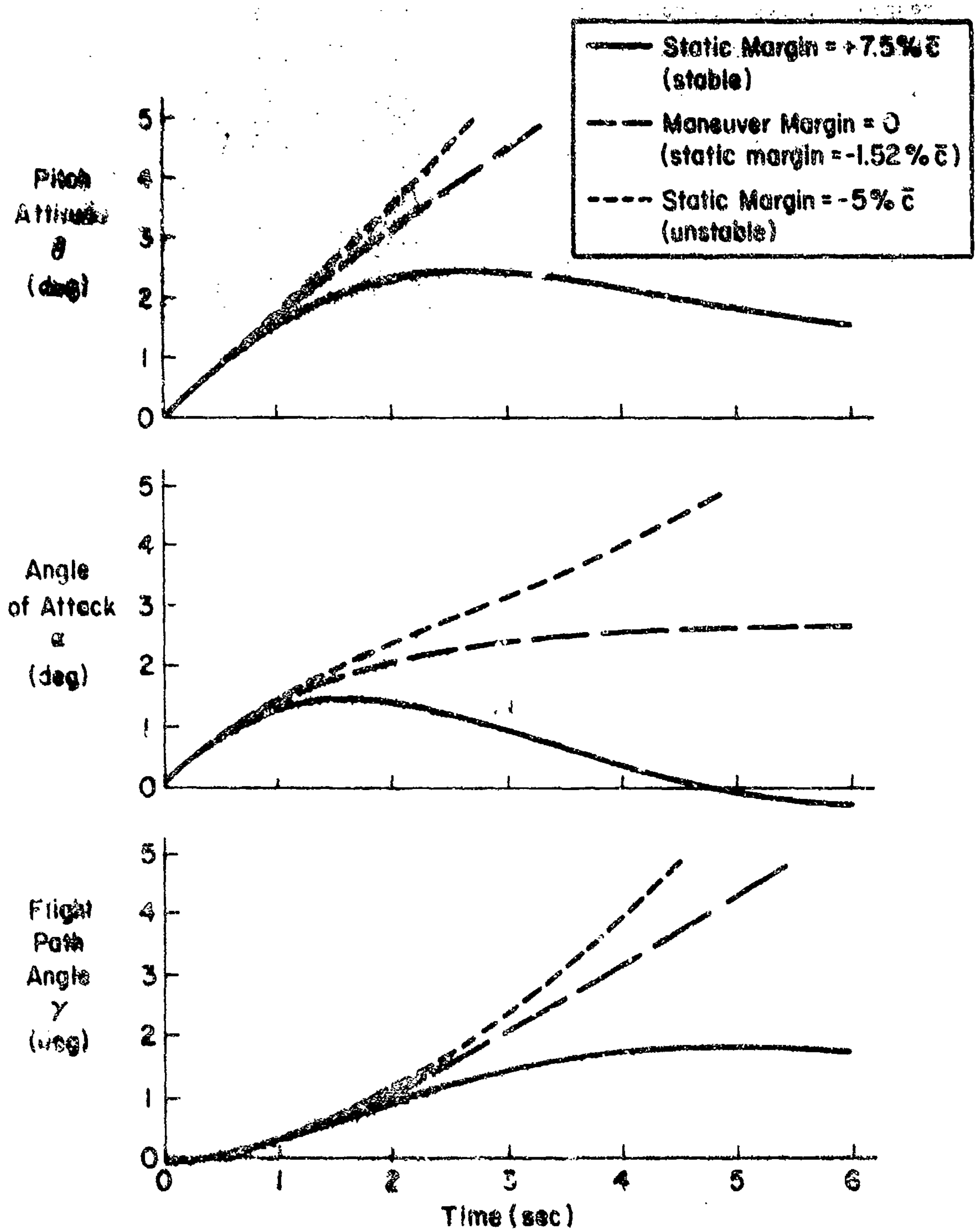


Figure 11. Generic RSS Transport Short-Period Motions for Cruise Conditions (M = 0.74, h = 36,000 ft); Input δ_H = -1 deg/sec Impulse

responses, regardless of static margin, are essentially indistinguishable for the first second, and that the negative margin case is close to the zero maneuver margin situation for another second or so.

Some appreciation of longer-time effects, where the phugoid mode is involved, can be obtained by considering the three-degree-of-freedom aircraft dynamics as indicated in Figure 12. The time scale here is contracted by a factor of 10, and that the short-period responses of Figure 11 correspond only to the first 5 or 6 seconds of the 1-1/2 minutes shown. The excitation for the Figure 11 responses is a 1/10 deg horizontal tail step. Also, the neutrally stable situation corresponds to zero static margin, as is appropriate for three-degree-of-freedom aircraft dynamics, rather than zero maneuver margin.*

The phugoid oscillation is plainly seen in the pitch attitude response for the 7.5 percent statically stable situation. This oscillation cannot be seen in the angle-of-attack response, because α is scarcely excited in the phugoid. On the other hand, the new trim angle of attack is seen very early, after just a few seconds. The zero static margin case shows a ramp in both angle of attack and pitch angle, as would be expected. This is the fundamental basis for setting static margin to near zero for reduction of trim drag. This indicates that the aircraft can be trimmed with very little steady-state surface deflection. Finally, the rapid divergence anticipated with the 5 percent negative static margin occurs just as it did for the short-period responses.

The generic aircraft data can also be used to illustrate a distinction between the aircraft dynamic characteristics based on short-period-alone considerations and those actually present when the full three degrees of freedom are taken into account. Starting with a stable static margin situation and reducing the stability (i.e., M_u becoming less negative to the zero static margin point and then becoming more

*Neutral stability for the three-degree-of-freedom aircraft dynamics occurs when $(Z_u M_w - M_u Z_w) = 0$. When M_u is zero, neutral stability is tantamount to $M_w = M_\alpha = 0$.

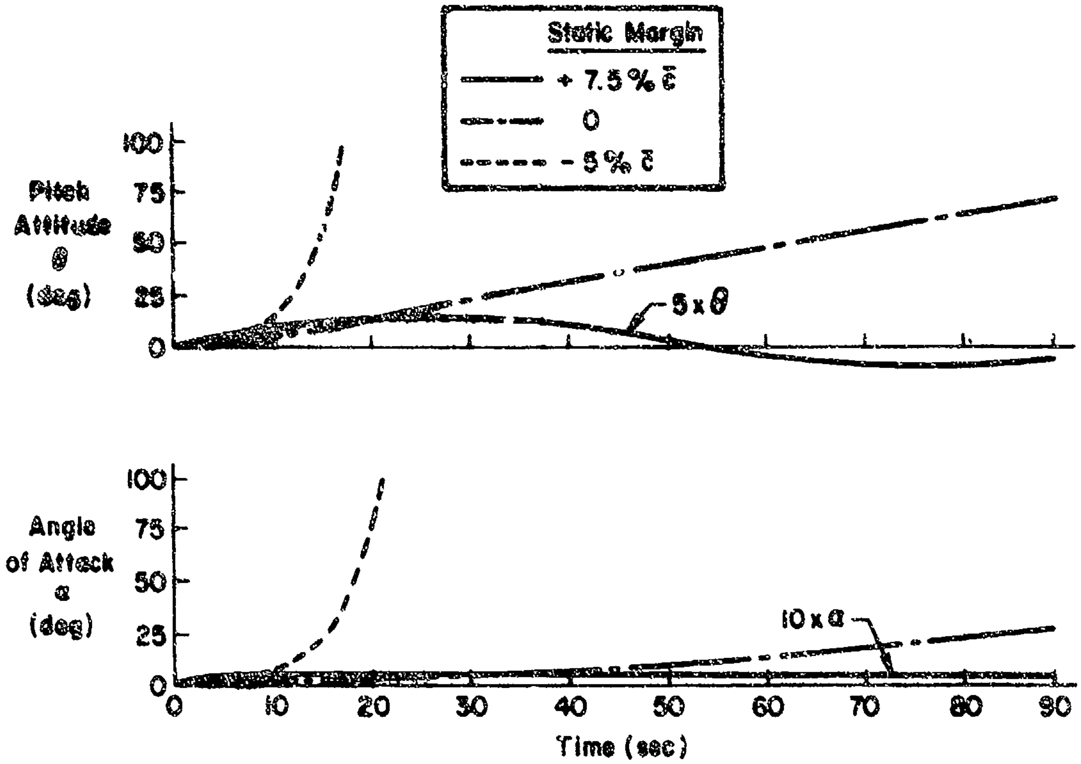
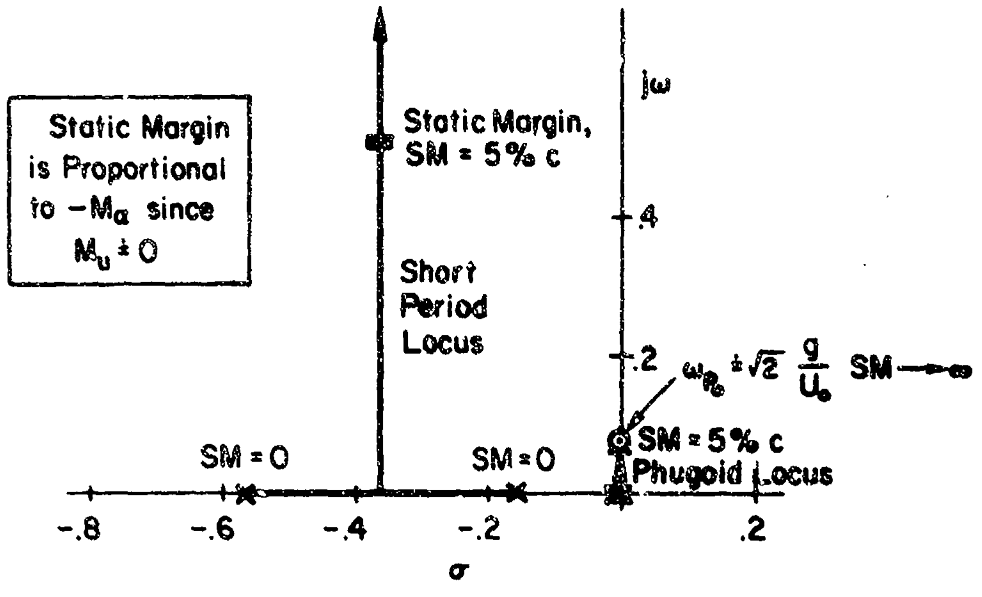
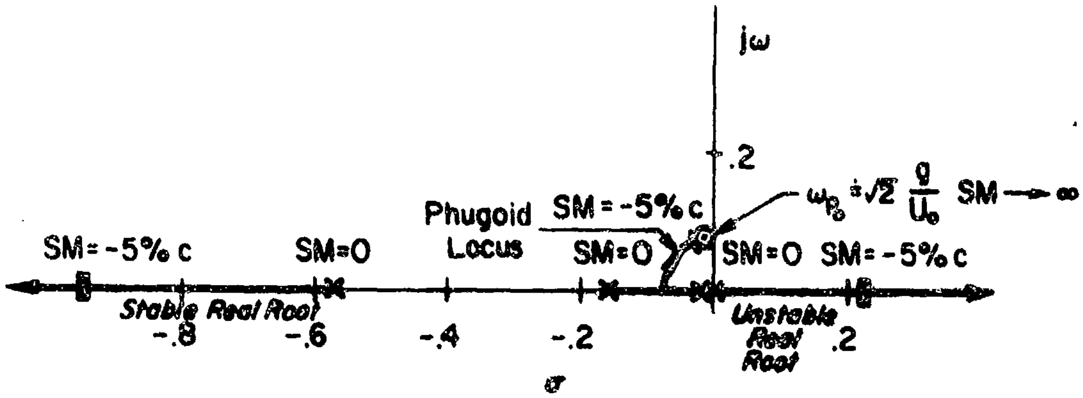


Figure 12. Generic RSS Transport Three-Degree-of-Freedom Responses for Cruise (M = 0.74, h = 38,000 ft); Input δ_H = -1/10 deg Step

positive), the actual locus of aircraft roots appears as shown in Figure 13. The short-period roots as they proceed toward the zero static margin condition initially appear just as described in connection with Figure 10. The phugoid mode, however, is also present in this root locus and is also affected by static margin shifts. What occurs there is an increase in phugoid damping until the phugoid mode becomes two real roots. One of these is actually at the origin when the static margin is zero, whereas the other is progressing toward one of the now real short-period roots. At the zero static margin condition, all four of the aircraft roots are real. As the stability is further reduced, the root at the origin proceeds into the right half plane; the high-frequency short-period root proceeds further into the left half plane; and the other two roots, one from the short period and one



a) Root Loci For Stable Static Margin Variation (M_α<0)



b) Root Loci For Unstable Static Margin Variation (M_α>0)

Figure 13. Effect of Static Margin Variations on Three-Degree-of-Freedom Conventional Airplane Modes

from the phugoid, approach each other, rendezvous, and become a quadratic pair. As the static stability is further decreased, this pair approaches the classical two-degree-of-freedom phugoid mode wherein $\omega_p \doteq \sqrt{gZ_u/U_0} \doteq \sqrt{2g/U_0}$. In this classic phugoid (Reference 22) the angle of attack is fixed while airspeed and pitch attitude oscillate, interchanging potential and kinetic energy, damped only by drag. Thus, the actual divergence in the three-degree-of-freedom case does not stem from the short period, but rather from the phugoid. The important point to be noted is that stability is just neutral in the three-degree-of-freedom motions when the static margin is reduced to zero. Use of the short-period approximation indicates neutral stability when the maneuver margin is zero. (In all of this discussion, the pitching moment change due to speed change, M_u, is assumed to be zero, so that static stability is governed entirely by M_α.)

Another interesting perspective about the short-period response can be gained using the peak q/q_{ss} versus $1/T_{\theta_2}\omega_{sp}$ coordinates of Figure 9 as a backdrop for variations in stability. In principle it might seem that almost any response is available (i.e., any point within the $0 < \zeta < 1$ space of Figure 9 could be reached) if one only designs and balances the aircraft configuration properly. This is partly true in that any desired short-period damping ratio, ζ_{sp} can be achieved using a pitch damping augmentor. But, because the path/attitude lag is a given for a particular wing configuration, the adjustment of c.g. (and hence of M_α) can lead only to a tightly constrained set of dynamic response properties. This is shown in Figure 14 for the Generic RSS aircraft in cruise. The curve shows what is attainable in terms of overshoot, damping ratio, etc. For instance, it indicates that very large static margins are accompanied by large overshoots induced by both the ζ_{sp} and $1/T_{\theta_2} - \omega_{sp}$ spread. This is to be expected when total short-period damping is constant and ω_{sp} is increased, as occurs when the c.g. is moved forward (see Figure 13a). As static stability is reduced due to decreasing M_α as the c.g. is moved aft, the pitch rate overshoot decreases and the damping ratio increases. The curves defining the attainable dynamic properties can be shifted, mainly up and down, by

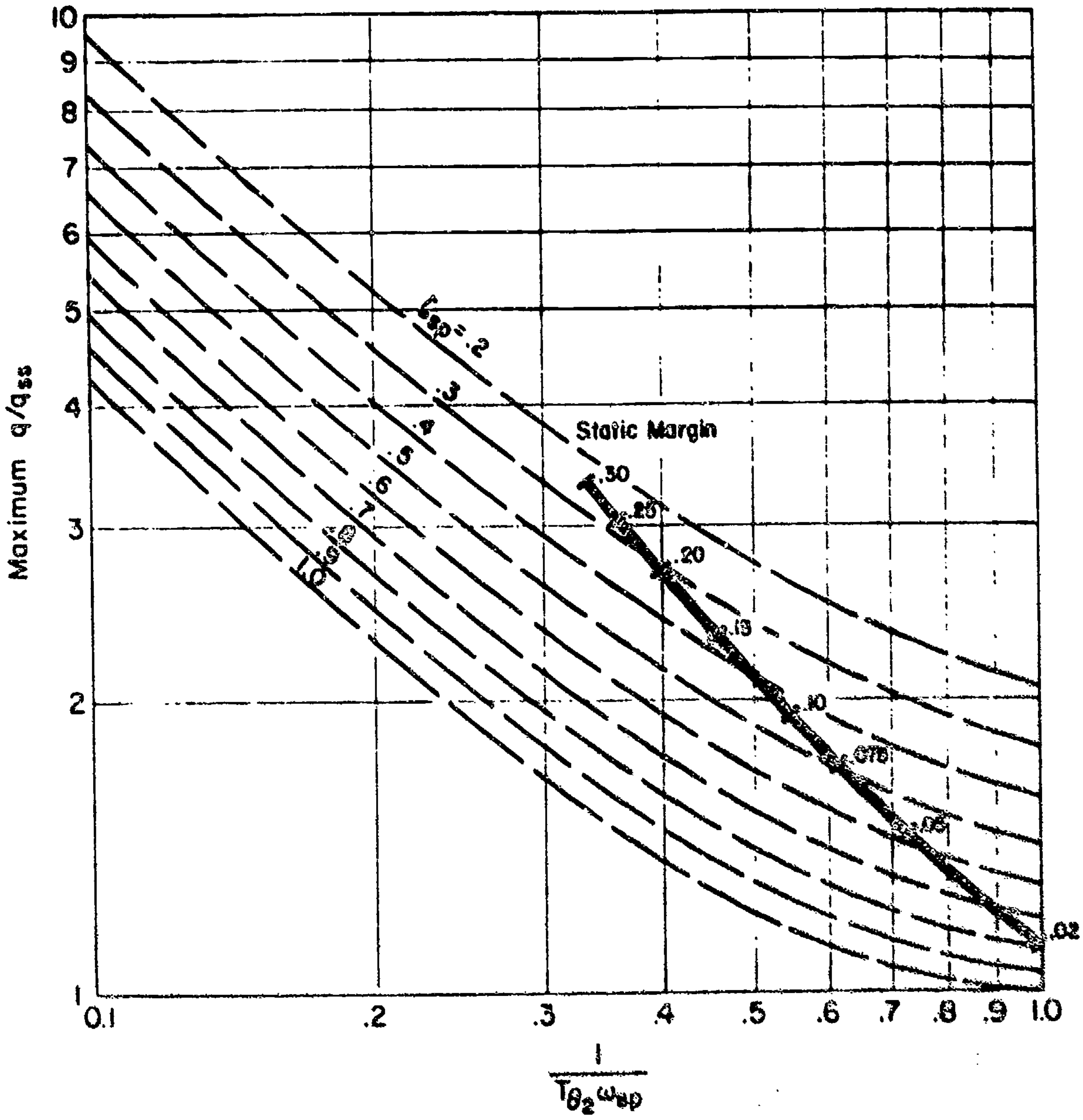


Figure 14. Pitch Overshoot Variation with Static Margin for Conventional Aircraft (Generic Aircraft in Cruise)

augmenting M_{α}, but the general trend will essentially parallel the one illustrated. There is thus only a narrow band about the curve shown which defines the possible short-period dynamics for this airplane.

E. A TYPICAL AUGMENTOR TO CORRECT RSS AIRCRAFT STABILITY DEFICIENCIES

As a price of the performance benefits which relaxed static stability aircraft enjoy they suffer from reduced short-period damping, low undamped natural frequency, and, perhaps, short-period divergence. A variety of full-authority augmentation systems can be constructed to correct these deficiences and, at the same time, significantly improve the aircraft flying qualities and reduce pilot workload. The augmentor must apply pitching moments to the aircraft which create or enhance particular stability derivatives. This can be accomplished by feeding back a variety of aircraft motion quantities. Some candidates are noted in Table 1. The most obvious stability derivatives to augment are those that cause the trouble in the first place, i.e., M_q and M_{α}, to improve damping and stability, respectively. A commonly used surrogate for pitching velocity is pitch attitude rate, $\dot{\theta}$, which is identical to the pitching velocity, q, in straight and level flight. The difference between q and $\dot{\theta}$ is most easily appreciated in connection with a constant altitude coordinated turn. In such a turn the steady-state pitch attitude rate is zero, whereas the steady-state pitching velocity, Q_o, is given by $Q_o = R_o \tan \Phi_o$ (where R_o is the yawing velocity and Φ_o is the bank angle). The short-period damping could also be augmented by a rate of change of angle-of-attack feedback, which would add to the natural $M_{\dot{\alpha}}$ of the airplane.

To improve the static stability, M_{α} can be augmented favorably by using the control to develop an additional pitching moment proportional to angle of attack. From the stability and pilot control standpoints either aerodynamic or inertial angle of attack will give the same result in principle, although the responses to aerodynamic disturbances will depend on which is used. Alternatives include the creation of a pitching moment due to pitch angle, M_{θ}, or its near equivalent, $M_{\int q}$, using

TABLE 1. SOME ELEMENTARY FEEDBACK CONTROL POSSIBILITIES TO CORRECT RSS AIRCRAFT STABILITY DEFICIENCIES

GENERAL EFFECT	PRIMARY EFFECTIVE STABILITY DERIVATIVE(S) AUGMENTED OR CREATED	FEEDBACK CONTROL POSSIBILITY
Improves Short Period Damping	$M_{\dot{\theta}}$	Pitch Attitude Rate $\dot{\theta} \rightarrow \delta$
	M_q	Pitching Velocity $q \rightarrow \delta$
	$M_{\dot{\alpha}}$	Angle of Attack Rate $\dot{\alpha} \rightarrow \delta$
Increase Static Stability	M_θ	Pitch Attitude $\theta \rightarrow \delta$
	$M_{\int q}$ (Same as M_θ when $\Phi = 0$)	Integral of Pitching Velocity $\int q \, dt \rightarrow \delta$
	$U_o M_{\int q}$	Integral of Normal Acceleration $\int a_z \, dt \rightarrow \delta$
	M_α	Angle of Attack $\alpha \rightarrow \delta$
	M_u	Speed $u \rightarrow \delta$

the integral of the pitching velocity, $\int q\,dt$. When one recalls that the normal acceleration is $a_z = \dot{w} - U_0 q$, a similar attitude-like corrective moment can be developed from the integral of the normal acceleration. Finally, the creation of a pitching moment due to speed changes by creating an M_u can also eliminate divergence and provide static stability.

Although all these and other possibilities are theoretically suitable for improving the modal damping and stability characteristics of the short period, all suffer some deficiency or other as the basis for a control system design. Problems of instrumentation and sensing, including biases and sensor excitation by disturbances, control system compensation needed for flight condition changes, etc., must enter into comparative consideration of practical systems. The transition from one flight phase to another, the effective dynamics as presented to the pilot, and the response of the augmented aircraft to external disturbances are also affected by the particular feedbacks chosen and must be considered in fundamental comparisons of candidate systems. These features and the consequences on effective aircraft dynamics of the various feedback control possibilities are treated in detail in Reference 22. Some additional topics of particular interest to RSS flying qualities will be described in the next section when some of the side effects of heavy augmentation are treated.

To permit the development of some flying quality issues for heavy augmentation we will use a typical example of an augmentor suitable for RSS aircraft. The issues drawn will, of course, pertain explicitly only to the example system, although some can be generalized for other system possibilities.

The flight control system selected for our example is shown in Figure 15. This system performs five functions, as follows:

- Creates a high degree of effective static stability for the augmented aircraft.
- Improves the damping of the effective short-period mode.

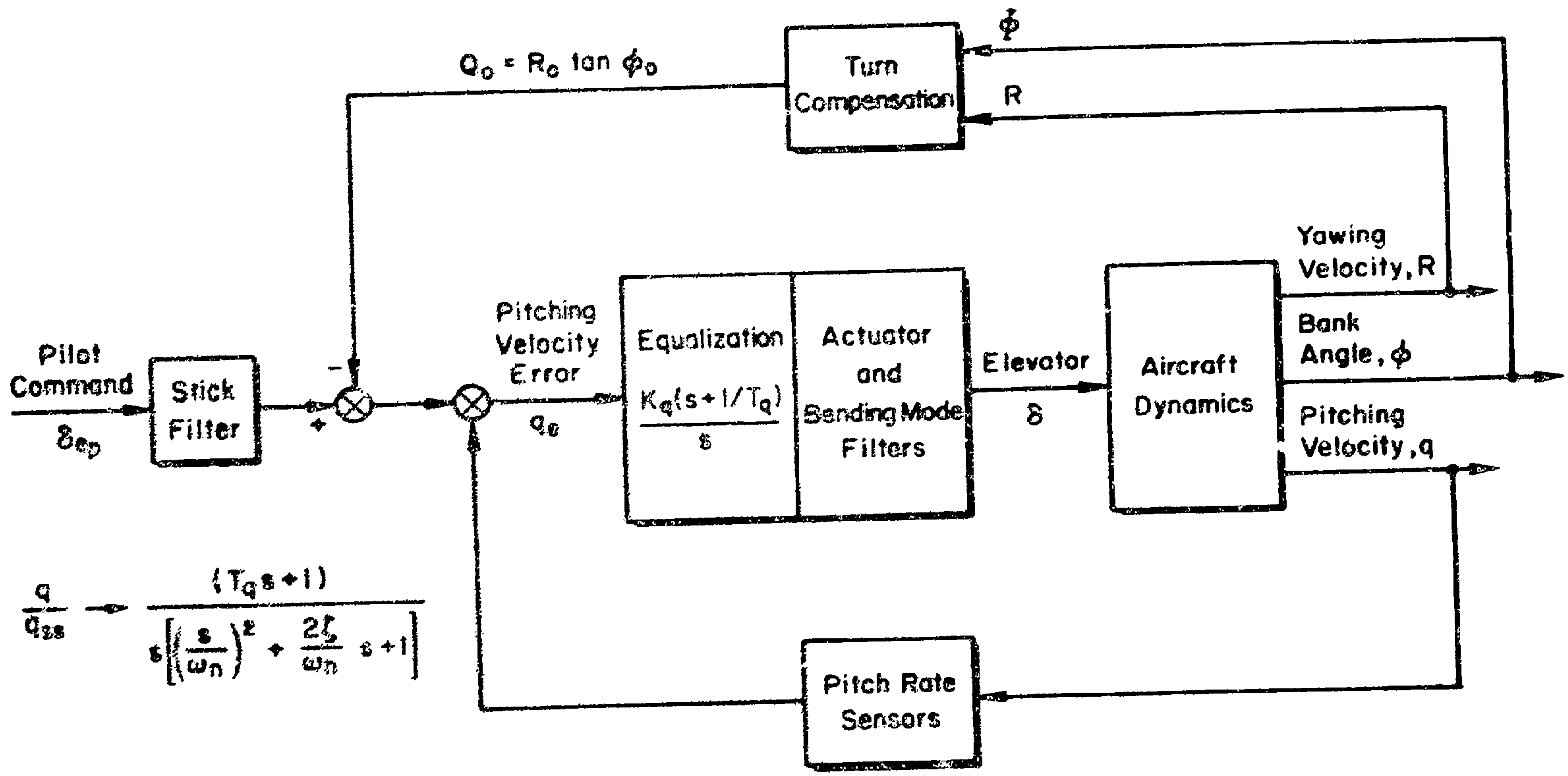


Figure 15. Typical Pitch SAS for Heavily Augmented Aircraft

- Provides a pitch rate command/attitude hold platform for piloted control.
- Regulates against external disturbances, with emphasis on pitch attitude maintenance rather than weathercocking.
- Provides automatic up-elevator compensation for turning flight.

As a flight-critical system, all the system elements except possibly those involved in turn compensation would be multiply redundant. This is one reason for basing the system on pitch rate sensors which are simple, hardy, relatively insensitive to bias errors, and easily made part of a minimum complexity multiple-redundant system. With skewed sensors, for instance, five rate gyros can provide dual fail-operate capability for rates in all three axes.

The basic low-frequency control law for the Figure 15 augmentor, which drives the elevator servo with a signal proportional to pitch rate error, q_e, and the integral of pitch rate error, q_e/s, is simply:

$$\delta = \underbrace{K_q q_e}_{\substack{\text{Propor-}\\ \text{tional}\\ \text{Term}}} + \underbrace{\frac{K_q/T_q}{s} q_e}_{\substack{\text{Inte-}\\ \text{gral}\\ \text{Term}}}$$

Note that the equation is just the block labeled "equalization" in Figure 15. Thus actuator and other higher-frequency dynamics are assumed to be negligible for the current discussion. This equation can be expressed in the time domain, noting that $q_e/s = \int q_e \, dt$, by

$$\delta = K_q q_e + \frac{K_q}{T_q} \int q_e \, dt$$

Consequently, the augmentor as a stabilizer creates a pitching moment proportional to $\int q \, dt$ and one proportional to q. (The pitch rate error is $q_e = q_c - q$; because the pitch rate command $q_c = 0$ when the augmentor

is acting only as a stabilizer, $q_e = -q$ in unattended regulation tasks.) When the aircraft-alone dynamics include a divergence, the aircraft/augmentor combination will be a conditionally stable system. That is, a minimum gain, K_q, is needed to rid the system of any divergence due to the aircraft-alone static instability. At the other extreme, the maximum gain possible is set by the closed-loop system stability limits. If the aircraft is considered only as a rigid body, these limits will depend primarily on the high-frequency lags due to the actuator, rate sensor, and other computational or filter dynamics within the closed-loop system. When aircraft flexible mode properties are also included, they too will also affect the closed-loop system stability and maximum gain. Depending upon the design approach taken, the flexible mode characteristics may be either phase or gain stabilized. In the latter case, filtering to attenuate the flexible mode will appear as part of the stabilizer loop. Lags from both the flexible modes and these filters will, in turn, affect the high-gain stability limit.

When the saturation characteristics of the aircraft control surface (and surface rates) are taken into account, the maximum gain may be further restricted. The higher the open-loop gain, K_q, of the augmentor, the smaller the pitching velocity error needed to saturate the control. If the aircraft alone has even some slight inherent stability, this may be of little consequence. However, when the aircraft-alone is divergent, a saturated control will not correct for this divergence except when the pitching velocity error is less than that corresponding to limit control conditions. The pilot command input is deliberately limited to avoid saturating the controls, but external disturbances are not. In fact, the possibility of control saturation due to shears and other atmospheric disturbances is one reason for the selection of the Figure 15 system. Some of the other stabilization possibilities listed in Table 1 result in systems which are not as tolerant to external disturbances and can cause significantly higher probabilities of limiting elevator positions.

Although this particular exemplary system has many advantages and has been used on a number of aircraft (e.g., the Space Shuttle Orbiter

and some modern fighters), it has its own peculiarities. For instance, the presence of the forward loop integrator, while needed for either a pitch rate or normal acceleration system to accomplish the static stability function, can be troublesome in some flight phases involving transient changes in trim conditions (e.g., takeoff) if not properly accounted for using appropriate synchonizers or flight-phase-tailored SAS functional modes. On the other hand, this system is the simplest one available which accomplishes the functions listed above. It also serves as an excellent paradigm and point of departure for developing the types of flying qualities issues involved in heavily augmented RSS aircraft.

Reference 24 uses the generic aircraft RSS transport characteristics as the basis for a control system example design. Cruise and approach conditions are examined for two versions of the Figure 15 pitch SAS (that is, with and without the integrator in the equalization). When the integrator is absent, the SAS is in essence a large authority, high-gain pitch damper, which has the effect of reducing but not completely eliminating any divergent tendency. This version also amounts to a pitch rate command/rate hold, rather than attitude hold per se; although it has quite similar regulation properties.

The exemplary SAS of principal interest here does contain the integrator and exhibits all of the features listed previously. The changes in the linear dynamics of the aircraft due to the control system can be illustrated using a system survey of the closed-loop system. This survey consists of a number of complementary root locus and frequency response plots. A typical set of plots for the cruise condition with a -5% $\bar{c}$ margin is given in Figure 16. [Other sets for approach and for both cruise and approach without the integrator are given in Reference 24. The key points can, however, all be made with the single example of Figure 16.]

The discussion below of Figure 16 will show that a properly designed augmentor can result in pitch attitude response characteristics which are identical in form to those of the aircraft short-period mode. The parameters governing the response, however, are <u>entirely different in</u>

$$-\frac{q}{q_e} = \frac{\overbrace{1.935\,(0.0065)}^{(1/T_{\theta_1})}\overbrace{(0.459)}^{(1/T_{\theta_2})}\overbrace{(2.0)}^{(1/T_q)}}{\underbrace{(-0.238)}_{(1/T_{sp_2})}\underbrace{(0.941)}_{(1/T_{sp_1})}\underbrace{[0.152\,,0.0781]}_{[\zeta_p,\,\omega_p]}}$$

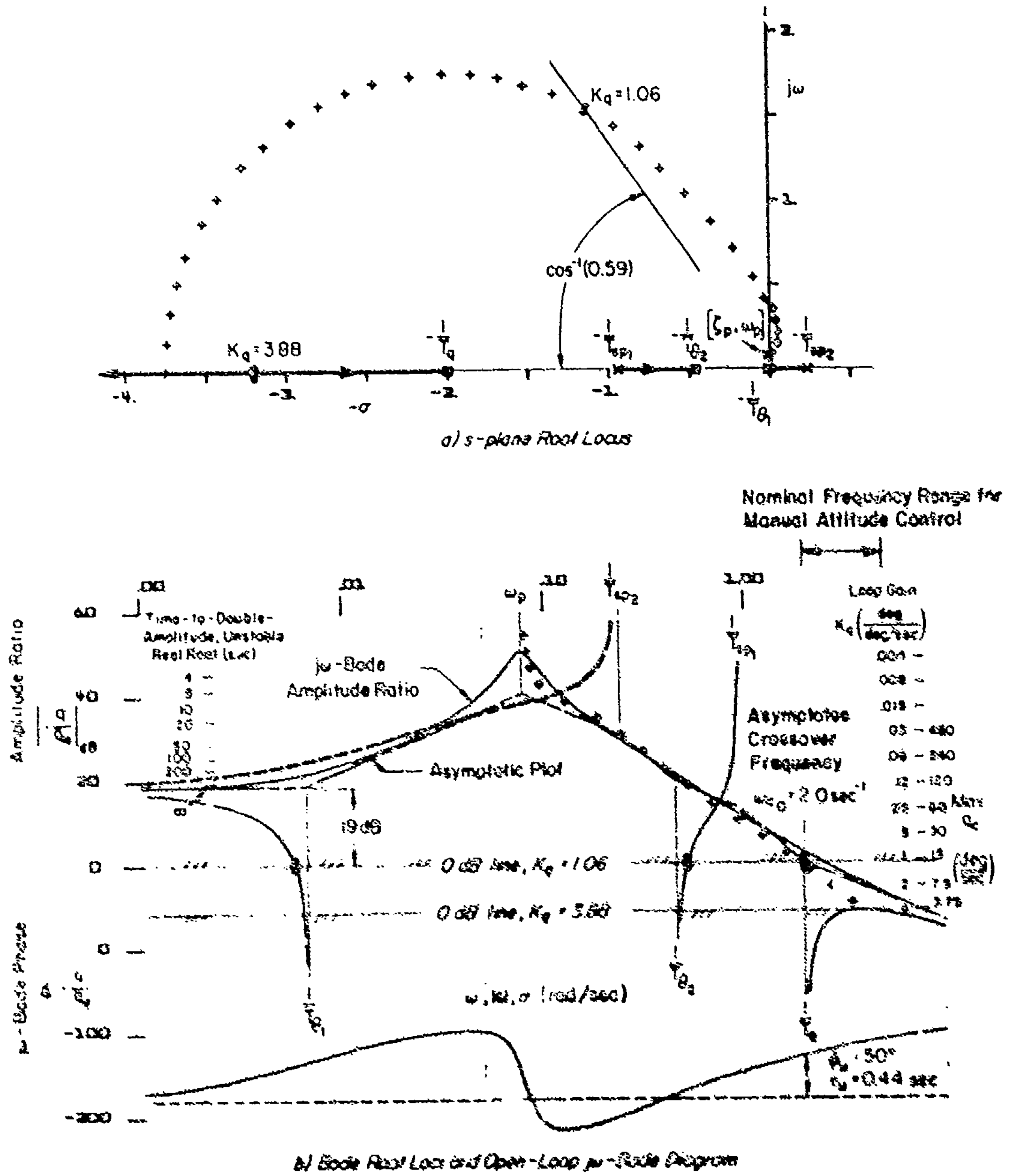


Figure 16. System Survey of Aircraft/Augmentation System Dynamics (Cruise, Static Margin = 5% $\bar{c}$; q, $\int q\,dt \rightarrow \delta_H$ Closure, $1/T_q = 2.0$)

their origin and may be significantly different in kind. Because of this, the aircraft pitch attitude response for the heavily augmented aircraft will bring new flying quality considerations to the fore. Because the demonstration of this point will involve a detailed consideration of the effects of closed-loop feedback control on the dynamics of the airplane, the explanations and some of the concepts may be unfamiliar to some. They may wish to skip to the end of this article where an expanded reiteration of the conclusions already stated above is given.

Figure 16a is a conventional s-plane root locus which shows the closed-loop roots of the system as control system gain, K_q, is increased. The starting points are the poles of the airplane located at $-1/T_{sp_2}$ (located in the right half of the s-plane, indicating its character as a divergence), $-1/T_{sp_1}$ (on the real axis in the left half plane), and the phugoid $[\zeta_p, \omega_p]$ (a lightly damped complex pair near the origin). The control system also has a pole at $s = 0$ due to the integrator. This is not shown on the plots because it exactly cancels a free s in the numerator of the airplane pitch rate due to elevator deflection, q/δ_h, transfer function. As the controller gain, K_q, is increased, the corrective moments applied to the airplane by the elevator modify the poles of the closed-loop system. Some of these at very high gain approach the zeros of the open-loop system. There are three open-loop zeros in the Figure 15 augmentor. Two are airplane characteristics, located at $-1/T_{\theta_1}$, in the left half plane but very near the origin, and at $-1/T_{\theta_2}$, well out the real axis in the left half plane. The control system lead time constant, T_q, is shown in Figure 16a as the zero at $-1/T_q$. As gain is increased the various airplane modes are modified as follows:

1) The airplane short-period divergence, $1/T_{sp_2}$, is decreased as gain, K_q, is increased; becomes stabilized as it passes through the $j\omega$-axis; and finally terminates on the airplane zero at $-1/T_{\theta_1}$ as gain approaches infinity.

2) The short-period subsidence, with time constant T_{sp_1}, proceeds along the real axis to the right toward $-1/T_{\theta_2}$. Part of the damping given up by this subsidence is transferred as an increase in damping to the divergence.

3) The phugoid, which for the airplane-alone is stable but lightly damped (ζ_p = 0.152), is initially driven toward instability as the augmentor gain is increased. The portion of the closed-loop phugoid locus in the right half plane is shown with ⊕⊕⊕. After a substantial gain increase (as will be shown below), the closed-loop phugoid hooks back into the left half plane and becomes stable. This stable part of the locus is shown as +++, and progresses along a nearly circular arc. This arc is nearly centered on the control system zero at $-1/T_q$. Also, the radius of the nearly circular segment is approximately $|1/T_q|$. As gain is increased further, the undamped natural frequency, damping, and damping ratio of this quadratic mode increases until ultimately $\zeta_p' = 1$* when the arc hits the real axis. The undamped natural frequency of this critically damped mode is just slightly less than $2/T_q$. As control gain is further increased, the quadratic pair then becomes overdamped, with one progressing out toward the left on the real axis while the other proceeds toward the right, ultimately ending on the control lead zero at $-1/T_q$.

From a controller design standpoint, the most important features to be appreciated from the s-plane root locus of Figure 16a are: 1) the correction of the divergence; and 2) the influence of the controller lead time constant, T_q, over the shape and magnitude of the locus of the complex roots $[\zeta_p', \omega_p']$.

The perambulations of the closed-loop roots as controller gain varies, while nicely depicted on the s-plane root locus, can also be shown with controller gain and associated quantities as an ordinate, rather than as a parameter along the plot. This is accomplished with the so-called Bode root locus. It uses a semi-log presentation for

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*The prime on ζ_p' indicates it is a closed-loop damping ratio of a quadratic mode which started, at zero gain, from the phugoid ζ_p, ω_p.

conventional open-loop system frequency response ($j\omega$-Bode plot of amplitude ratio and phase) and Bode root locus constructions (root loci with Bode format), wherein gain is an independent variable on the plot. Figure 16b presents these several diagrams. To begin with, the ordinate is an amplitude ratio plot of the open-loop transfer function expressed in decibels. (For a gain, K, which is normally expressed in linear units, the value in dB is 20 $\log_{10}$ K. Thus, when K = 1 the value in dB is zero, while when K = 10 or 0.10, the corresponding dB values are +20 dB and −20 dB.) The abscissa, which is a logarithmic scale, is a generalized frequency. When the plot is a conventional Bode frequency response plot, this frequency is ω in rad/sec. Complex components of the root locus show undamped natural frequencies, $|s|$ or ω_n, in rad/sec. Finally, real roots when plotted here are also in units of 1/sec. Thus, interpretation of the abscissa depends on the particular curve, and is either ω, $|s|$, or σ.

The conventional $j\omega$ Bode diagram consists of separate amplitude ratio and phase plots for the total open-loop transfer function with $s = j\omega$. The amplitude ratio and phase angles are shown in Figure 16b. The amplitude ratio of the frequency response borders on and is approximated by an asymptotic plot made up of straight lines with slopes which are integer values of ±20 dB per decade. This asymptotic plot has its breakpoints occurring at values of frequency equal to magnitudes of the open-loop poles and zeros. Thus, the asymptotic plot has a slope of zero from 0 frequency to a frequency of $1/T_{\theta_1}$, whereupon it breaks up with a +20 dB/decade slope. This continues until the frequency $\omega = \omega_p$, where the phugoid, being a second order, causes a break from +20 dB/decade to −20 dB/decade. The next breakpoint occurs at the magnitude of the divergence, $|1/T_{sp_2}|$ when the net slope of the asymptotic amplitude ratio plot becomes −40 dB/decade. At the airplane short-period lead, $1/T_{\theta_2}$, the slope changes again to −20 dB/decade and continues until the short-period subsidence at $1/T_{sp_1}$ is encountered. After this the amplitude ratio slope goes back to −40 dB/decade until the controller lead breakpoint at $1/T_q$, whereupon the asymptotic plot slope shifts and progresses on at −20 dB/decade. The regular $j\omega$ frequency response is quite

close to this asymptotic plot except at the breakpoints themselves. It departs from first-order breakpoints by 3 dB, while the departure from second-order breakpoints is equal to $[1/2\zeta]_{dB}$. Thus the smaller the damping ratio, the more extreme the departure of the actual $j\omega$-Bode from the asymptotic Bode plot. This gives rise to the resonant peaks associated with highly oscillatory modes as seen in frequency responses. In Figure 16b the phugoid is a prominent example.

If now the root locus shown on the s-plane of Figure 16a is presented in these logarithmic coordinates, the variation of roots with open loop system or controller gains is illustrated directly. The several branches shown on the Bode root loci of Figure 16b have symbols which correspond with those on the s-plane root locus. These branches can be described as follows:

1) The locus from the short-period subsidence, $1/T_{sp_1}$, goes directly toward the numerator lead at $1/T_{\theta_2}$. At very low gains the change due to closing the loop is very small, and similarly, at high gains $1/T_{sp_1} \doteq 1/T_{\theta_2}$. Only in the intermediate region does the root migrate toward the zero fairly rapidly with gain.

2) The complex root starting at the phugoid, with undamped natural frequency ω_p when the open-loop gain is very small, proceeds along the locus shown by +++ and ◇◇◇ symbols. Much of this progress occurs with a nearly constant slope of approximately −40 dB/decade. The closed-loop oscillatory mode becomes unstable at quite low gains. (See the controller linear gain scale at the right of the plot). For controller gains $0.004 < K_q < 0.06$ deg/deg/sec the closed-loop phugoid is unstable. This is the portion of the locus depicted with ◇◇◇. For gains greater than this the phugoid is stable and the circular arc of Figure 16a corresponds to the nearly constant slope of 40 dB/decade on the Bode root locus in Figure 16b. The real axis is encountered by this locus for a gain of

approximately $K_q = 3.08$, after which the complex roots divide into two real roots. One, as is shown on both the Figure 16a and b plots, goes into the $1/T_q$ control system lead, while the other progresses indefinitely out the high-frequency asymptote.

3) The divergence is shown as a dashed locus, starting with $1/T_{sp_2}$ at very low gains and proceeding leftward across the Bode root locus. When the gain is $K_q = 0.12$ (19 dB less than the reference $K_q = 1.06$ condition) this mode is neutrally stable. It, for higher gains, then drives further into the left half plane, shown here as a heavy solid line progressing toward $1/T_{\theta_1}$.

In a good control system design, the system reflects several different response and stability considerations. These include:

- Responses which are similar to those of low-order, well-damped, rapidly responding systems. (This implies that the low-frequency open-loop poles are, in their closed-loop manifestations, driven nearly into open-loop zeros, such that they nearly cancel. Examples on Figure 16b for the $K_q = 1.06$ reference 0 dB line are the close proximity of the lead at $1/T_{\theta_1}$ with the closed-loop pole, shown as ■, stemming from the divergence, and a similar proximity of the lead at $1/T_{\theta_2}$ to the pole arising from the short-period subsidence.

- Insensitivity of the response to gain changes. This is illustrated by the nearly vertical slopes of the loci driving into $1/T_{\theta_1}$ and $1/T_{\theta_2}$ around the reference crossover region. Because the slopes are so steep small changes in gain, or for that matter small changes in the open-loop aerodynamics which change $1/T_{\theta_1}$ and $1/T_{\theta_2}$, will not materially affect the near cancellation of these

closed-loop leads and lags. Thus they will hardly affect any changes in the response.

- System stability with large stability margins. In the present example of the conditionally stable system, a margin of 19 dB exists on the low-gain end relative to the reference $K_q = 1.06$. Thus, nearly a factor of 10 in gain reduction would be needed to get back to the divergence. At the high-frequency end, the crossover of $\omega_c = 2$ rad/sec (which incidentally sets the desired controller gain at $K_q = 1.06$) is consistent with a phase margin, ϕ_M, of 50 deg and a delay margin, τ_M, of 0.44 sec. Thus, high-frequency lags or parameter uncertainties currently ignored in the design would have to contribute 50 deg of phase lag, or a pure time delay of 0.44 sec, before the closed-loop system would be neutrally stable at the gain selected.

- Well damped and rapidly responding closed-loop system dominant mode(s) to resist and thereby reduce the effects of disturbances. In the present example the dominant mode is the oscillation stemming from the phugoid which, for the nominal controlled gain, has a damping ratio of 0.59 and an undamped natural frequency of 1.92 rad/sec. This mode is therefore both stiff and well damped.

- The closed-loop system bandwidth should be large relative to the pilot control latencies so that the augmented airplane is responsive to pilot command and requires little pilot anticipation or compensation for precision control. In the present case the crossover frequency of 2 rad/sec and the other resulting modal characteristics provide a very spritely response (as will be illustrated further later).

- The control system gain should be low enough so that the augmentor is very seldom saturated. Saturation may be viewed as reducing the gain and thus progressing from the nominal 0 dB line closed-loop roots back toward those of the open-loop aircraft. When completely saturated, the effective controller gain approaches zero and the effective aircraft dynamics are those of the airplane alone. Unfortunately, in this event the pilot also has no control available in one direction, since the surfaces are saturated. These kinds of considerations are easy to show on the plot of Figure 16b as commanded pitching velocities which would just saturate the system when gains are set at particular levels. Scales showing the maximum pitching velocity commandable without saturation, q_c max, is given on the right side of the Bode root locus plot next to that for the controller gain, K_q, in linear units. Another useful scale for partial saturation and other low-gain operations is given on the far left side. This shows the time to double amplitude of the divergent root at the several gain levels (in essence this is a crossplot of the locus from $1/T_{sp_2}$ toward the $1/T_{\theta_1}$ lead). For instance, a divergence with 6 sec time to double amplitude corresponds to a gain K_q of 0.010 deg/deg/sec. This is an enormous reduction in effective loop gain (factor of 388) in terms of saturation or other gain reduction phenomena, showing that the system is extremely robust at the nominal value of K_q = 1.06.

The description above, even though of a summary overview character, may appear tedious and complicated. The net effect, however, of a well-tempered design is remarkably straightforward. It is that the closed-loop pitching velocity response to a step pilot input will have the approximate form:

$$\frac{q}{q_{ss}} \doteq \frac{(T_q s + 1)}{s[(s/\omega_n)^2 + (2\zeta/\omega_n)s + 1]}$$

The ζ and ω_n in the denominator are those of the closed-loop oscillatory mode which progressed from the phugoid, while the lead time constant, T_q, is the controller lead. The approximation takes advantage of the fact that the short-period subsidence and divergence are both modified by the effects of feedback control and, for favored gains, do not even appear in the pitching velocity response since they are nearly cancelled by the airplane leads at $1/T_{\theta_1}$ and $1/T_{\theta_2}$. Feedback control has thus provided us with a low-order effective system as far as pitch attitude response to pilot inputs is concerned. It is in fact identical in form to that of the short period of the airplane alone. This is especially apparent in the time response for the total system shown in Figure 17.

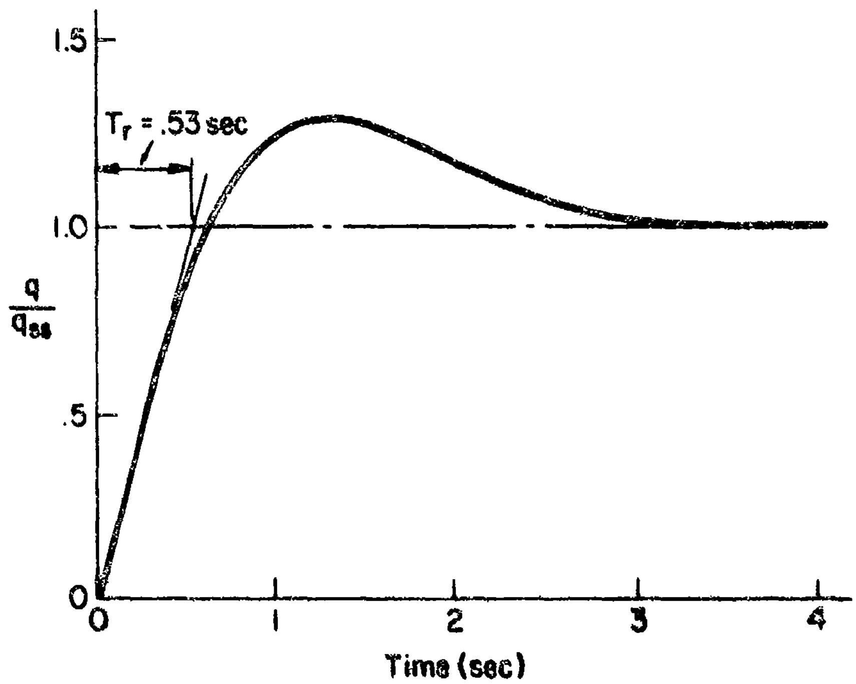


Figure 17. Pitch Rate Response to Step Pitch Rate Command for the Augmented Aircraft in Cruise Flight

There the characteristic initial ramp-like, nearly constant pitch acceleration rise with the overshoot and subsequent return looks very similar to the exemplary time response of Figure 8 and to that for the generic aircraft in Figure 11 for the 7.5 % $\bar{c}$ static margin. The closed-loop system response is more rapid in rise time and achieves a steady state in less time.

There is an important difference between the time responses of Figure 17 and those of Figure 11 in that the closed-loop response for Figure 17 is that for <u>all time</u> rather than just the first few seconds. In other words, the phugoid, which was illustrated in Figure 12 for the conventional aircraft, is no longer present as a long period oscillation. Instead, in the exemplary closed-loop system, it has been markedly increased in frequency and damping and is now, in fact, the single oscillatory mode present. (Calling this extremely well damped, high natural frequency mode a "closed-loop phugoid" is based only on its origin. For all intents and purposes it is, of course, a short-period mode, and henceforth will be treated and described as such.)

The net conclusion of this article are that: provision of feedback control on a statically stable, relatively poorly damped, relaxed static stability aircraft can result in a set of vehicle dynamics which are substantially identical in form to those of the short-period alone; and more importantly, while the pitch attitude form may be the same, the parameters are entirely different in kind.

I. COMPARISON OF PITCH ATTITUDE RESPONSE GOVERNING PARAMETERS FOR CONVENTIONAL AND SUPERAUGMENTED AIRCRAFT

In the exemplary case study summarized in the last section, the pitch attitude response lead, T_q, came from the augmentor lead rather than from the airplane T_{θ_2}. In fact, the airplane pitch attitude lead, T_{θ_2}, is not even present in the response, since it was essentially canceled by the closed-loop subsidence originating at $1/T_{sp_1}$. Also, the undamped natural frequency and damping, while superficially similar to the short period, depend instead primarily upon the control system lead

time constant, T_q, which set the circular arc along which the ζ_p', ω_p' closed-loop roots proceeded, and the total open loop gain, $K_q M_\delta$, which located ζ_p', ω_p' at a given spot along this arc. These parameters, in fact, are those associated predominantly with the highest frequency and next highest frequency asymptotes of the Figure 16b Bode diagram. Using just these two high frequency asymptotes as an approximation for the total system, it is easy to show that the approximate values of ω_n and ζ are given by:

$$\omega_n^2 = \frac{K_q M_\delta}{T_q}$$

$$= \frac{\omega_{ca}}{T_q}$$

and

$$\zeta = \frac{1}{2}\sqrt{K_q T_q M_\delta}$$

$$= \frac{1}{2}\sqrt{T_q \omega_{ca}}$$

Also, the approximate normalized rise time is

$$\omega_r T_r = \frac{1}{T_q \omega_n}$$

$$= \frac{1}{\sqrt{T_q \omega_{ca}}}$$

Here the quantity $\omega_{ca} = K_q M_\delta$ is the crossover frequency of the 0 dB line with the high-frequency asymptote (see Figure 16b). The subscript "a" distinguishes it from the crossover frequency ω_c, which is based on the value where the open-loop $j\omega$-Bode amplitude ratio crosses the 0 dB line. These crossover frequencies are perhaps the most important single parameters of closed-loop controls, since they divide the world of inputs and responses into two different categories. At frequencies well below the crossover frequency, errors due to disturbances are suppressed and

command inputs are followed. At higher frequencies the feedback control action gets progressively weaker, so that command following and disturbance suppression is reduced. The crossover frequency is often used as a surrogate for the system bandwidth, which has the same physical meaning noted above. Interestingly enough, for the closed-loop control system the rise time is just the inverse of ω_{ca} (i.e. $T_r = 1/\omega_{ca}$).

For augmented relaxed static stability aircraft which have the general type of characteristic wherein the airplane pitch attitude lead T_{θ_2} no longer appears in the effective augmented aircraft dynamics, we have emphasized that the pitch attitude characteristics are different in kind but not in form from those of a conventional aircraft. This particular type of limiting case heavily augmented aircraft we shall refer to as "superaugmented". The superaugmented distinction is made to highlight the differences — that superaugmented aircraft have attitude characteristics which depend primarily on the crossover frequency $\omega_{c_a} = K_q M_\delta$, and the controller lead T_q, and that these characteristics may differ in kind from those of even fairly heavily augmented conventional craft. In this sense, superaugmented aircraft are an idealization which may not be always approached by heavily augmented aircraft. On the other hand, the Space Shuttle is an ideal example of a superaugmented aircraft, for $1/T_{\theta_2}$ is completely suppressed in its pitch attitude response and is replaced by a control system lead. Superaugmentation is a useful concept because it is sufficiently idealized to simplify the drawing of issues and understanding, while not so far removed from reality as to make the considerations academic.

With this prelude, we can now turn to a comparison of the key pitch attitude response parameters for conventional aircraft and for superaugmented aircraft. This is accomplished in Table 2.

Table 2 summarizes the attitude lead and quadratic (effective short period) mode characteristics for a conventional aircraft short period and for an idealized superaugmented aircraft.

For the conventional airplane, covered at the left, the table reiterates yet again that the attitude lead and short-period undamped

TABLE 2
COMPARISON OF PITCH ATTITUDE RESPONSE GOVERNING PARAMETERS FOR
CONVENTIONAL AND SUPERAUGMENTED AIRCRAFT

DYNAMIC PROPERTY	CONVENTIONAL		SUPERAUGMENTED	
	PARAMETERS	PRIMARY DESIGN FACTORS	PARAMETERS	PRIMARY DESIGN FACTORS
Lead Time Constant T	$\frac{1}{T_{\theta_2}} \doteq -Z_w + \frac{Z_\delta}{M_\delta} M_w$	Set Predominantly by Airframe: C_{L_α}; wing	T_q	Governed predominantly by closed-loop control system Stability, Response, and Margins **Control System Parameters** T_q - Lead time constant K_q - Gain **Airframe Parameter** M_δ - Surface effectiveness (C_{m_δ})
Undamped Natural Frequency ω_n	$\omega_{sp}^2 = Z_w M_q - M_\alpha$	Maneuver Margin; $C_{M_{CL}}$, Tail	$\omega_n^2 = \frac{\omega_{c_a}}{T_q}$ $= \frac{K_q}{T_q} [M_\delta]$	
Normalized Rise Time $\omega_n T_r$	$\frac{1}{T_{\theta_2} \omega_{sp}}$		$\frac{1}{T_q \omega_n} = \frac{1}{\sqrt{T_q \omega_{c_a}}}$ $T_r = \frac{1}{\omega_{c_a}}$	
Damping Ratio ζ	$(\zeta\omega)_{sp} = -(Z_w + M_q + M_{\dot{\alpha}})$	C_{L_α}; Wing C_{m_q}; Tail Pitch Damper	$\zeta = \frac{\sqrt{T_q \omega_{c_a}}}{2}$ $= \frac{1}{2}\sqrt{K_q T_q [M_\delta]}$	
Delay Time τ_d		Actuator and Manual Control lags		Actuator lag + Stick filters + Bending Mode filters + Computational delays

natural frequency, and hence the rise time, depend primarily on aircraft configuration characteristics and the way the aircraft is balanced. The damping ratio also is predominantly a function of configuration, although a pitch damper can provide a good deal of design latitude.

For the superaugmented aircraft, the discussion of the last article emphasized the relative lack of sensitivity to aircraft configuration characteristics and the relative importance of the controller properties as they affect the closed-loop aircraft/augmenter system. The primary design factors described there were considerations of a closed-loop character, including the system stability, response, bandwidth, stability margins, etc. The most important composite factor underlying the dynamics of the superaugmented vehicle is the crossover frequency (of the asymptote), ω_{c_a}. This quantity, given by

$$\omega_{c_a} = K_q M_\delta \left(1 + \frac{Z_\delta}{M_\delta} M_{\dot{w}}\right)$$

$$\doteq K_q M_\delta$$

is an indication of:

- The total system gain comprising both controller (K_q) and aircraft control effectiveness (M_δ) parameters.
- The system bandwidth, which indicates the frequency range of good command following and disturbance suppression.
- The rapidity of system response, i.e., rise time $T_r \doteq 1/\omega_{c_a}$.
- The system damping ratio, in that ω_{c_a} is a key factor (together with the controller lead time constant, T_q) in setting the damping ratio, ζ.

The first three properties of the crossover frequency listed above are qualitatively applicable to all feedback control systems which have a low-pass closed-loop character. (Low pass here means that at frequencies up to the bandwidth the output follows the input quite well, whereas at higher frequencies the output will drop off in amplitude relative to the input — thus the low frequencies are "passed" through

the system while frequencies higher than the bandwidth are attenuated or "not passed.") The fourth property is a special one for superaugmented systems which share the specific characteristics of the example case. It is one reflection of the idealized superaugmented situation wherein only two parameters, the attitude lead (T_q) and crossover frequency (ω_{c_a}), define all the system input/output characteristics except the overall response scaling between output and input.

Another manifestation of the two-parameter character of the idealized superaugmented aircraft can be seen in connection with the maximum pitch rate overshoot versus $1/T_q\omega_n$. This is shown in Figure 18. The possible variation in damping ratio, overshoot, and normalized rise time is encompassed by the straight line superimposed on the now familiar background plot. Notice that for a normalized rise time of 1, the damping ratio is 0.5 and the undamped natural frequency is $1/T_q$. At the other end is a normalized rise time of 0.5, accompanied by a $\zeta = 1$ and an $\omega_n = 2/T_q$. Any system between these two extremes has excellent closed-loop control, system stability, and margins. Again the parameters which set the actual location on the attainable line are the crossover frequency, ω_{c_a}, and the control system lead time constant, T_q.

It is instructive to compare the pitch overshoot variation with static margin of Figure 14 for conventional aircraft with the Figure 18 law of overshoot for superaugmented airplanes. In the one case the permitted variable is the way the aircraft is balanced, i.e., the static margin, whereas in the other the $\omega_{c_a}T_q$ product provides the variation. The first thing to notice is that the trends are in somewhat different directions relative to the background constant damping ratio (ζ) coordinates. For the conventional aircraft, increased static margin has a concomitant increase in the undamped natural frequency and decrease in the normalized rise time. This aspect is similar to that for ω_n and normalized rise time, $1/T_q\omega_n$, for the superaugmented aircraft. On the other hand, the damping ratio of the short period decreases and the overshoot increases for the conventional aircraft, while the opposite trend is present for the superaugmented airplane.

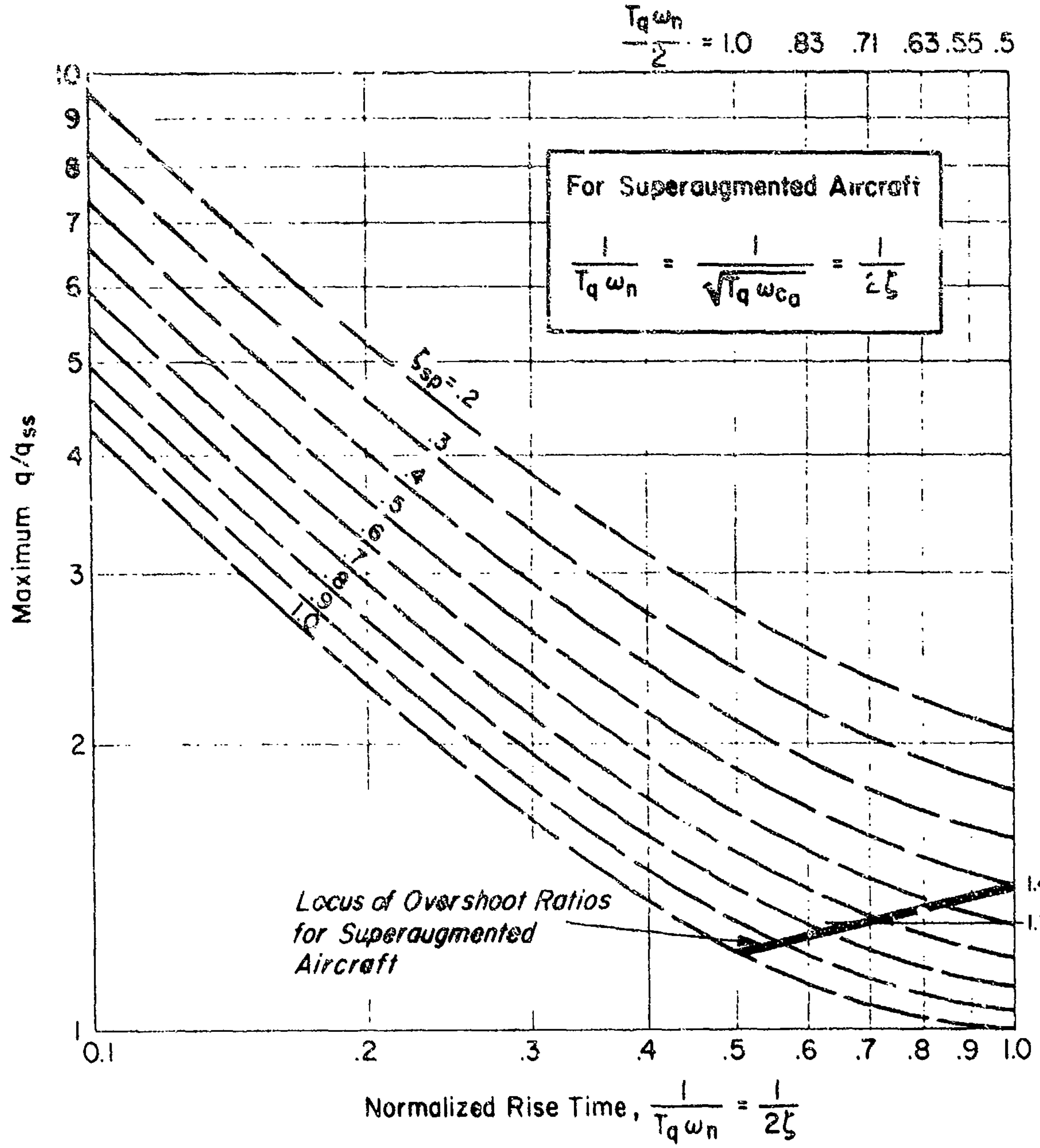


Figure 18. Maximum Pitch Rate Overshoot Variation for Superaugmented Aircraft

A major distinction can also be made between the superaugmented and conventional aircraft with reference to the aerodynamic characteristics which underlie their responses. For the conventional aircraft, even in the short period, the stability derivatives Z_w, M_q, and $M_{\dot{\alpha}}$ together with their variations with flight condition, are major governing parameters. When the complete three-degree-of-freedom airplane characteristics are also taken into account, several more derivatives become important (e.g., Z_u, M_u, X_w, etc.). On the other hand, to the extent that the augmentation system can be made to approach the superaugmented characteristics, the aerodynamic parameters of importance reduce to the surface effectiveness, M_δ. Potential variations in other derivatives must, of course, be assessed in the design process to assure that no possible variation could upset this applecart, but in actual system operation the primary sensitivity and variations of interest are those of M_δ. In some ways, this sparsity of airplane-characteristic-dependence for aircraft which approach the superaugmented state offers a major advantage. The system which provides superaugmentation will itself be complex in that it is multiply redundant, yet the properties of any single channel of the multiple redundant system are extremely simple, straightforward, and sensitive to only a very few parameters. Thus, the concept of a "simplex" multiple redundant augmentor has some appeal and bears further consideration.

Finally, the ultimate comparison of the conventional and superaugmented vehicles is connected with the closed-loop precision path control flying quality aspects. Referring to Figure 5, we can now indicate why the augmented aircraft pitch dynamics block was not made more specific in terms of the subscripts for the quantities in the transfer function incorporated there. The attitude lead is now no longer T_{θ_2}, but the control system lead T_q, while the undamped natural frequency and damping ratio are unrelated to those of the conventional short period. Thus, the augmented aircraft pitch attitude dynamics are potentially fundamentally different than those of a conventionally augmented aircraft. Not the least important of these differences is the replacement of the T_{θ_2} lead by T_q, for now the attitude lead is not the same as the path/

attitude response lag (which is T_{θ_2} for both the conventional and superaugmented situations). On existing superaugmented vehicles, there is often a substantial difference between these two properties. For instance, on the Shuttle Orbiter, in a typical approach flight condition the value of $1/T_{\theta_2}$ is about 0.54 sec^{-1} whereas $1/T_q$ is 1.5 sec^{-1} (Reference 25). Similarly, for the generic RSS transport in cruise $1/T_{\theta_2}$ is 0.46 sec^{-1} while $1/T_q$ is 2 sec^{-1}. This difference between the attitude/path lag and the effective lead in pitch attitude, as well as the potential differences in the basic high-frequency response mode, may be of fundamental importance in flying qualities and flying qualities research. The importance on the positive side could occur because the response characteristics in pitch attitude of the idealized superaugmented aircraft offer many potential benefits. The listing of functions (associated with Figure 15) provided by heavy augmentation is an example of some of these. There are also negative aspects, some associated with flying qualities criteria and others with the side effects introduced by augmentor system. These topics will be described in the next section.

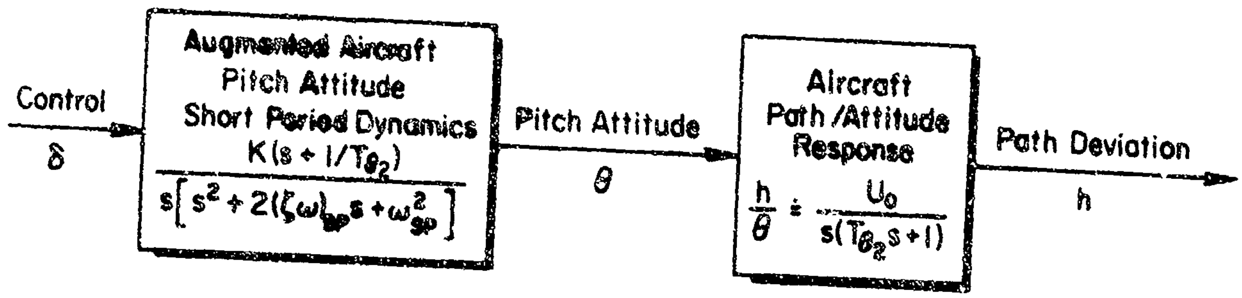
SECTION IV

FUNDAMENTAL AND MECHANIZATIONAL SIDE EFFECT FLYING QUALITY CONSEQUENCES OF HEAVILY AUGMENTED AIRCRAFT

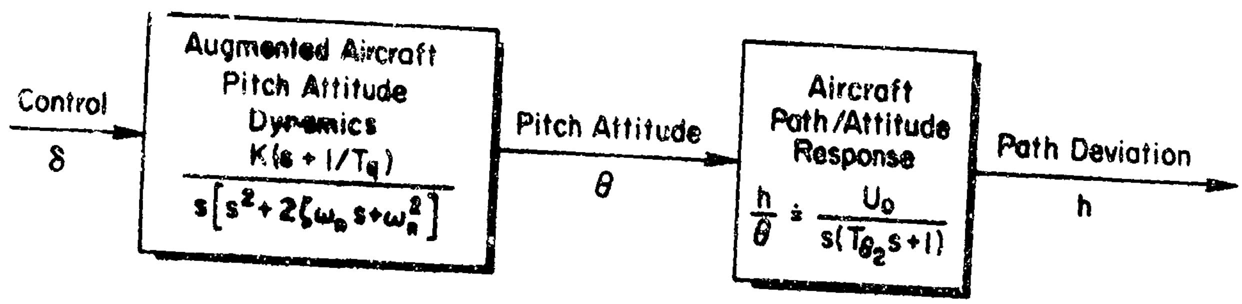
A. FUNDAMENTAL FLYING QUALITY CONSEQUENCES FOR SUPERAUGMENTED AIRCRAFT

Much of the last section was devoted to a development of the similarities and differences in precision path control for different categories of effective airplane dynamics. Figure 19 shows a simplified comparison of conventional and heavily augmented aircraft dynamics. The latter represents the closed-loop dynamics of the aircraft plus augmentor when idealized at the superaugmented extreme. The two blocks for each case constitute the effective aircraft dynamics portion of the total precision path control system of Figure 5. As demonstrated in the last section and remarked many times, the distinction between conventional and superaugmented closed loop aircraft/augmentation system dynamics is present in the pitch attitude characteristics block. Although the forms are the same, the parameters are different in both the numerator lead and the denominator quadratic which describes the aircraft's high frequency (short-time) attitude response characteristics. In review, the key distinctions made in the last chapter are:

- The aircraft path/attitude response, h/θ, is the same for both conventional and superaugmented aircraft;
- The augmented aircraft pitch attitude short-term characteristics differ in that:
  1) The lead T_{θ_2} for the conventional aircraft is the same as the path/attitude lag whereas the lead for the superaugmented aircraft T_q may be quite different from T_{θ_2}.
  2) The undamped natural frequency and damping of the effective short-period mode for the conventional aircraft depends primarily on aircraft flight condition, weathercock stability, and pitch damping (sometimes augmented).



a) Conventional Aircraft



b) Superaugmented Aircraft

Figure 19. Effective Aircraft Dynamics (Aircraft + Augmentation System) for Conventional and Superaugmented Aircraft

3) The undamped natural frequency and damping for the superaugmented aircraft depends predominantly on the augmentation system (lead and gain) and aircraft control effectiveness (M_δ) parameters.

- The low frequency and trim characteristics for the conventional aircraft are not reflected by the short-period attitude dynamics approximation, whereas the super augmented aircraft pitch attitude dynamics are appropriate for low frequency and trim.

With these differences now established as at least idealizations, the key question is what effect, if any, do they have on flying qualities and safety? Unfortunately, the answer to this question is not yet in. Specific research addressed to these problems is needed on both a generalized, i.e., for transports as a class, and ad hoc basis specific to a particular transport configuration. What we shall attempt below is a resume of current status on these issues.

At the outset it should be recognized that almost all of the criteria and the very great preponderance of flying quality research data which underlie the various flying quality criteria were obtained on aircraft in the conventional category. The flying quality data base for heavily augmented and superaugmented aircraft is exceedingly sparse. In fact, much of the available data base and even some of the relevant criteria in the existing or proposed Military Specification (References 17 and 30) cannot be used directly for superaugmented aircraft. A recently completed study in which all available criteria were considered for a particular superaugmented aircraft — the Space Shuttle Orbiter — showed that they were sometimes inapplicable or gave very ambiguous and confusing results (Reference 25).

One of the pioneering attempts to specify the flying qualities of a relaxed static stability vehicle derives from the Space Shuttle Orbiter. Because this vehicle under piloted control is always an "effective vehicle" which inherently intermingles airframe-alone with some augmentation, this specification from the outset considered only the aircraft/augmentor closed-loop system characteristics. The Shuttle flight control system is flight critical, so no attention was paid to aircraft-alone dynamics in the requirements. The specifications also relate

directly to the pitch attitude control, with no specific mention to path control features. The statements are given in two parts. The first part is qualitative and states that the system shall provide a pitch rate output proportional to pilot inputs. This is accomplished using an augmentation system which is equivalent to the one we have been using in our examples (Figure 15). The second part is more quantitative in character and provides limits on the maximum and minimum steady-state pitch rates that can be commanded and a time domain boundary specification for transient responses. Only the latter is of interest to us here. The original subsonic pitch rate response boundaries, from Reference 26, are shown in Figure 20. These boundaries have shifted somewhat during the Shuttle's development, with the present (Reference 27) set also shown in Figure 20.

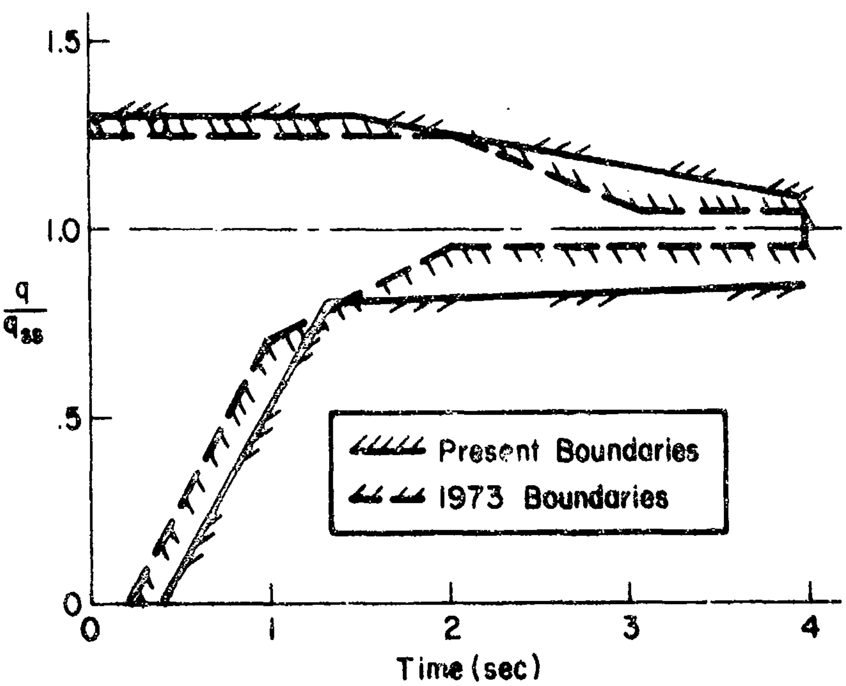


Figure 20. Exemplary Time Domain Response Boundaries for an RSS Aircraft

Perhaps the most constraining characteristic of these boundaries is the overshoot limit at 1.3. As can be seen by examining Figure 18, this maximum pitch rate overshoot for an idealized superaugmented aircraft requires a minimum damping ratio greater than $\zeta = 0.7$. Further, because of the constrained connections between the variables for our idealized superaugmented configurations, the maximum normalized rise time of the pitching velocity response would be less than 0.7. Also from Figure 18, using the upper scale, $T_q\ \omega_n/2$ would have to be greater than 0.71 or so. All of these are needed to live within the max q/q_{ss} overshoot boundary of 1.3. For a typical case, if we assume a rise time of 1 second to stay well within the lower bound, then the crossover frequency, ω_{c_θ}, will be 1 rad/sec. Then using the above cited exemplary numerical values and the formulas of Table 2, the control system (and effective aircraft pitch attitude) lead time constant T_q will be approximately 2 sec. The dynamics of the quadratic mode will then be $\zeta \doteq 0.7$ and $\omega_n \doteq 0.7$ rad/sec.

Transient responses for approach and cruise flight conditions of the generic RSS transport (Appendix A and Reference 24) used in this study are given in Figure 21. These responses are generally within the time response boundaries shown in Figure 20. The cruise response overshoot is essentially on the upper boundary. As already demonstrated by the simple numerical example above, this overshoot boundary will be a critical and highly constraining factor on any aircraft which has effective dynamics approaching those of the idealized superaugmented configuration.

While we hold no thesis for the pitch rate time response boundaries of Figure 20, they do appear to encompass the responses for our generic RSS aircraft and are generally compatible with the Shuttle itself, which is also an RSS aircraft. The boundaries can thus at least be considered exemplary, and can be used as a convenient framework from which to consider possible distinctions between conventional and heavily augmented aircraft. We shall use them in this strawman role below to serve as a backdrop for data comparisons.

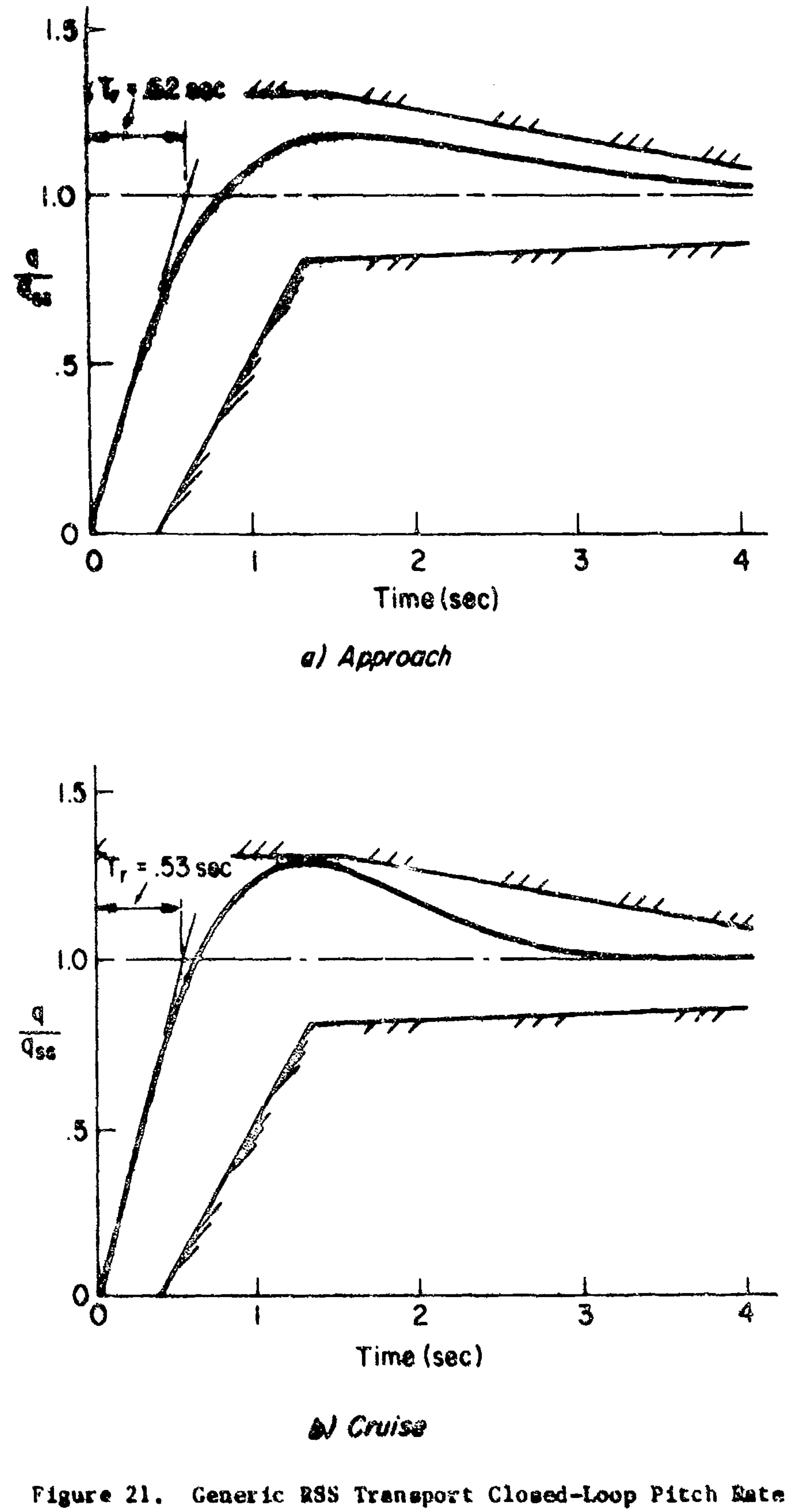


Figure 21. Generic RSS Transport Closed-Loop Pitch Rate Responses to Step Inputs

The best and most current set of flight data for conventional aircraft precision path control is probably the landing and approach higher order system (LAHOS) study of Reference 28. These experiments used the Calspan variable stability NT-33 aircraft in an approach and landing task. The evaluation flights were continued through touchdown. A large number of configurations were evaluated, usually by two pilots, with repeat evaluations being made randomly for many of the configurations. We have selected six of these configurations, summarized in Table 3, as being particularly relevant to the points at issue here. Three of the six selected LAHOS configurations exceeded the exemplary time domain RSS aircraft boundaries, and three had responses within those boundaries. Both sets are shown in Figure 22. With a pilot rating of 3-1/2 as a boundary between desirable and undesirable workload, corresponding approximately to the MIL-Spec Level 1 flying quality category, Figure 22a indicates that conventional configurations with good flying qualities may not meet the exemplary RSS boundaries. In fact, the overshoots are higher than would be allowed by the exemplary boundaries while the damping ratios are often considerably less. On the other hand, the three configurations which have responses which fall within the exemplary boundaries have overall pilot ratings between 6 and 7. It will be recalled that pilot ratings of 6.5 correspond to conditions where adequate performance cannot be attained with a tolerable pilot workload. These configurations, therefore, demand excessive workload and pilot compensation for control purposes. They are all poor from the flying qualities standpoint, and border on the unsafe by virtue of the poor control and high workload. Incidently, the flight path/attitude lag for both good and bad configurations is $1/T_{\theta_2} = 0.714\ \text{sec}^{-1}$. Since this feature is the same for all, the rating differences are probably associated with the attitude component of precision path control.

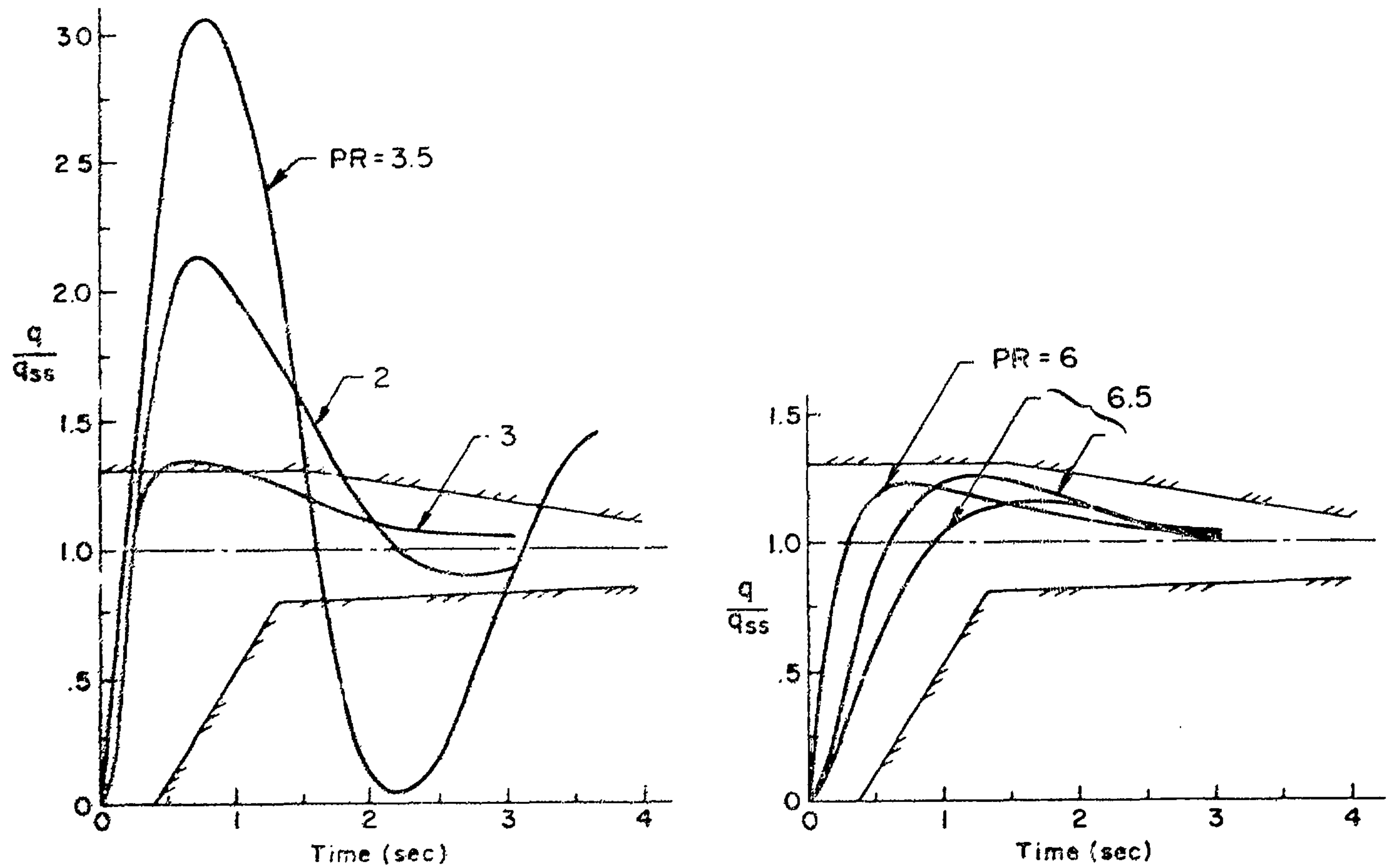
Thus, the LAHOS data indicate that <u>conventional aircraft which do meet the exemplary boundaries can have poor flying qualities whereas those which do not can have good flying qualities</u>. Indeed, the marked similarity between some of those responses shown in Figure 22b and the responses of the generic RSS heavily augmented aircraft in Figure 21

TABLE 3

SUMMARY OF SELECTED CONFIGURATIONS FROM REFERENCE 28

CONFIGURATION NUMBER (Ref. 28)	NORMALIZED PITCH ATTITUDE TO STICK FORCE TRANSFER FUNCTION	OVERALL COOPER-HARPER PILOT RATING		AVERAGE
		PILOT A	PILOT B	
CONFIGURATIONS OUTSIDE EXEMPLARY BOUNDARIES				
2-1	$\frac{7.41(.714)}{s[.57,\ 2.3]}$	2	2	2
3-C	$\frac{13.6(.714)(5.0)}{s(10.)[.25,\ 2.2]}$	2	5	3.5
4-C	$\frac{11.2(.714)(5.0)}{s(10.)[1.06,\ 2.0]}$	3	3	3
CONFIGURATIONS INSIDE EXEMPLARY BOUNDARIES				
4-0	$\frac{6.19(.714)}{s(1.08)(4.09)}$	6	—	6
4-3	$\frac{22.4(.714)}{s(1.42)(2.82)(4.0)}$	5.7	8	6.5
4-4	$\frac{11.2(.714)}{s(1.42)(2.0)(2.82)}$	7	6	6.5

Notation: $(1/T) = s + 1/T$, $[\zeta,\omega] = s^2 + 2\zeta\omega s + \omega^2$



a) Good Conventional Configurations Which do not Meet Exemplary Pitch Rate Requirement

b) Poor Conventional Configurations Which Satisfy the Exemplary Pitch Rate Requirement

Figure 22. Comparison of Conventional Configurations with Exemplary Tim Domain Boundaries for RSS Aircraft

could be used to support a contention that the flying qualities of the generic RSS aircraft would be poor. [Recognize that logically, however, all that can be said to this point is that the exemplary boundaries are not suitable to define good flying qualities for conventional aircraft.]

There is, of course, another interpretation. This is quite simply that, as we have emphasized all along, superaugmented aircraft are different in their characteristics and are not appropriately judged by data from conventional configurations. With this interpretation, the exemplary boundaries or something similar could conceivably still encompass the responses of heavily or superaugmented aircraft which have good flying qualities. Unfortunately, there are very few data available which apply to heavily augmented aircraft as considered herein and even less data which are pertinent to the idealized superaugmented condition. A very recent study, however, does contain one data point which does indeed approach the superaugmented situation. This appears in Reference 29, which examined the handling qualities of large airplanes in the approach and landing phase using the USAF-AFWAL/Calspan Total In-Flight Simulator. The study simulated a 1 million pound statically unstable airplane as the RSS baseline vehicle. Several control systems used to stabilize the aircraft were examined. Among these was one which corresponded to the augmentor of Figure 15. One of the cases studied had sufficiently high control system gain to approach the superaugmentation idealization. This response is shown in Figure 23. The overshoot, while less than the maximum in the exemplary boundary is extended somewhat further and the rise time is also fairly large. Nonetheless, this pitch rate response is not too far removed from the exemplary boundaries. This configuration was evaluated by both evaluation pilots used in the study and received generally good ratings. In its second evaluation by one pilot, in fact, it was given a Cooper-Harper rating of 1 which is extremely unusual (the same pilot initially evaluated it as 4). The pilot commentary indicates initial problems in trim, basically in attempting to "Keep the airspeed and attitude organized." After familiarization, however, the same pilot noted that "Airspeed control is excellent. Once I get it trimmed up it virtually holds the airspeed,

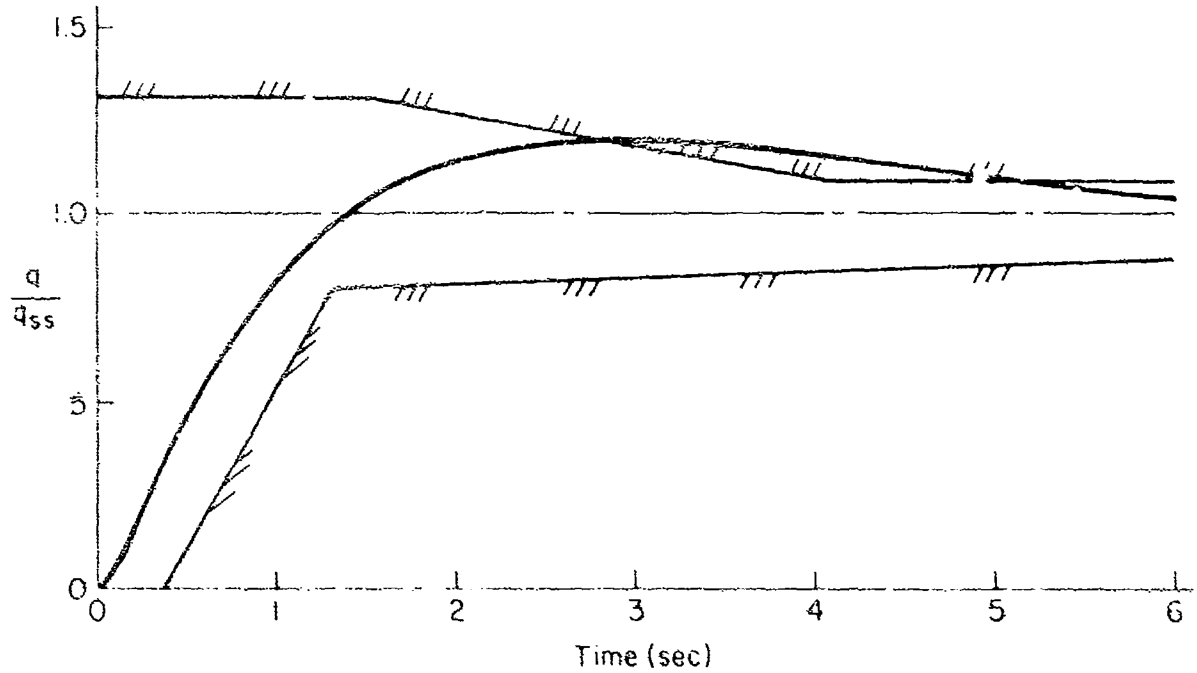


Figure 23. A Simulated Superaugmented Airplane Response for Approach and Landing (Ref. 29); $1/T_{\theta_2} = 0.53\ \mathrm{sec}^{-1}$, $1/T_q = 1\ \mathrm{sec}^{-1}$

holds attitude, and stays trimmed in turns." The other pilot indicated that "airspeed control was good, predictable." His summary comment was "No major problems, an excellent airplane." From these comments it would appear that in precision path control, a superaugmented configuration may indeed exhibit good flying qualities. There does appear to be a potential familiarization problem, although this is rapidly overcome. This one data point goes a long way toward justifying a position that heavily augmented RSS aircraft, especially as they approach the superaugmented condition, cannot satisfactorily be judged by criteria or compared with data from conventional aircraft. In other words, the distinctions between the conventional and heavily to superaugmented aircraft developed in the last section constitute major flying qualities issues which have yet to be resolved. In summary, to this point:

- The flying qualities of superaugmented and conventional aircraft are indeed different.

- Aircraft using the example augmentation system can possess good precision path control flying characteristics.

- The limits and sensitivity on flying qualities for heavily augmented aircraft are not yet defined.

B. PILOT FORCE VARIATION WITH SPEED CONSIDERATIONS

Among the most fundamental provisions of the FARs for stability and control of Part 25 aircraft are those called out in Sections 25.171 (General) and 25.173 (Static Longitudinal Stability). These, in essence, require that

> "The airplane must be longitudinally...stable..."
>
> "A pull must be required to obtain and maintain speeds below the specified trim speed, and a push must be required to obtain and maintain speeds above the specified trim speed."
>
> "The airspeed must return to within 10 percent of the original trim speed for the climb, approach, and landing conditions...and to within 7.5 percent of the original trim speed for the cruising condition...when the control force is slowly released from any speed within the range specified..."
>
> "The average gradient of the stable slope of the stick force vs. speed curve may not be less than 1 lb for each 6 kts".

A rigid constructionist reading of the FARs would be that no aperiodic divergences are permitted and that stick force per mile per hour must have a stable gradient. For a conventional aircraft (with fully-powered surface actuators and some control system friction so that stick-free and stick-fixed characteristics are the same), these statements are equivalent. This can be seen by considering the steady state angle of attack, speed, and attitude transfer functions for an elevator input. When the effects of pitching moments due to speed change associated with the stability derivative M_u are ignored, these steady-state transfer functions are given by:

$$\frac{\alpha}{\delta} = -\frac{M_\delta}{M_\alpha} = \frac{-C_{m_\delta}}{C_{m_\alpha}}$$

$$\frac{u}{\delta} = \frac{-M_\delta(1/T_{\theta_2})}{Z_u M_\alpha}$$

$$= \frac{U_o^2}{2g}\left(\frac{C_{m_\delta}}{C_{m_\alpha}}\right)\left(\frac{1}{T_{\theta_2}}\right)$$

$$\frac{\theta}{\delta} = \frac{X_u}{g}\left(\frac{u}{\delta}\right) + \frac{X_\alpha}{g}\left(\frac{\alpha}{\delta}\right)$$

$$= \frac{1}{g}\left(\frac{-C_{m_\delta}}{C_{m_\alpha}}\right)\left[X_\alpha - X_V \frac{U_o^2}{2g}\left(\frac{1}{T_{\theta_2}}\right)\right]$$

The term in brackets associated with θ/δ is normally positive. These equations can be interpreted as follows. For a conventional, statically stable aircraft ($C_{m_\alpha} < 0$) with stable phugoid mode, an up elevator command (negative elevator) will ultimately yield a positive angle of attack, α, a positive pitch angle, θ (nose up), and a negative speed change, u (slow down). Because the aircraft is statically stable there will be no aperiodic divergences. When the sign of M_α is changed, all these trends reverse. Accompanying these changes will be an aperiodic instability introduced by the negative static margin.

The above relationships can be put into another useful form by considering ratios of speed to angle of attack and attitude to angle of attack for elevator control inputs only. These are given by

$$\frac{u}{\alpha} = -\frac{U_o^2}{2g}\left(\frac{1}{T_{\theta_2}}\right)$$

$$\frac{g\theta}{\alpha} = X_\alpha + X_u\left(\frac{u}{\alpha}\right)$$

As can be seen, these ratios are substantially independent of the static margin and weathercock stability parameter, M_α. They are, therefore, essentially invariant with the degree of relaxed static stability.

If the control surface deflection, δ, in the above expressions is approximately proportional to stick force then the expression for u/δ can be taken as a surrogate for the inverse of stick force per unit speed change. It will be recalled that the inverse of the path angle lag is given approximately by the heave damping, $-Z_w$, which itself varies directly with speed. Consequently, the speed change per unit surface deflection will vary as the cube of the trim speed. Because the stick force per unit speed change is proportional to the inverse of u/δ, the stick force gradient with speed becomes very flat at high speeds. This can make it quite difficult to meet the numerical value of 1 lb for each 6 kts when the speed gets high enough.

The simple relationships indicated above lend some insight to the original construction of the FARs and at least some of their potential meaning. The requirement for a positive stick force gradient with speed change for conventional aircraft fundamentally assures the absence of an aperiodic divergence, and also permits trimability in both a short and long term framework. There is probably no simpler measurement to make in flight than stick force as a function of trim speed, so the statement of the requirement in these terms permits a straightforward test sequence to assure compliance.

When transonic effects enter, the pitching moment per unit speed change (proportional to M_u) cannot be ignored. The condition for neutral stability is then no longer $M_w = M_\alpha = 0$; but, instead $Z_u M_w - M_u Z_w = 0$. If, however, the stick force gradient is stable, all the other trends mentioned above still apply.

Let us turn now to a relaxed static stability aircraft equipped with a heavy augmentation system of the type shown in Figure 15 or its equivalent. For this aircraft/augmentor combination, the closed-loop system will be stable and will exhibit no aperiodic divergences. The basic system, however, is one which is rate-command/attitude hold, so that the pilot command δ_{e_p} gives rise to a pitch rate (even in the steady state) rather than to a (steady state) pitch attitude as in the conventional aircraft. The control system when not activated by the pilot thus fundamentally maintains the airplane trim in attitude, rather

than in angle of attack (or its surrogate, speed). The stick force gradient with speed is, in fact, zero. As noted in connection with the pilot comments for the simulated superaugmented airplane configuration of Figure 23, these effective vehicle dynamics were favorably considered. Remarkably, no comment was made directly connected with the neutral gradient of the stick force per kt. Indirectly, however, the initial familiarization required to "keep the airspeed and attitude organized" was noted.

There are few topics in aircraft stability and control and flying qualities that can generate more heated debate (and, conversely, less enlightenment) than the need for a positive as opposed to neutral stick force gradient with speed. As should be apparent from our introductory discussion above this is a bootless issue on conventional aircraft flown with a margin of static stability. Yet, even here, the gradient inevitably approaches zero as speed is increased. For heavily augmented or superaugmented aircraft with the exemplary system of Figure 15 or an equivalent augmentation, the issue becomes very important, for speed stability is inherently neutral as seen from the pilot's control point, δe_p.

There are many simulations and flight experiment results with augmentation systems akin to that of Figure 15 wherein the neutral speed stability has not been an important issue when contrasted with the very favorable features provided by the rate command/attitude hold augmentation (e.g. References 31, 35). The need for positive stick force stability with pitch rate command attitude hold systems was specifically addressed in the flight tests of References 35, 36. The conclusion was that there were no clear advantages to positive over neutral speed stability, at least when the aircraft was operated at the bottom or front side of the thrust required versus speed curve. The fundamental <u>attitude</u> stability, as opposed to weathercock stability, is ordinarily very favorable in terminal operations and other conditions wherein atmospheric disturbances can seriously affect precision path control. Also, the rate command/attitude hold properties of such systems are ordinarily viewed with favor. On the other hand, the pilot technique appears to

require some initial familiarization, especially in landing. The initial tendency, which is rapidly corrected by one or two practice landings, is to land long.

There is another feature of rate command/attitude hold systems which has received some pilot comment. Consider, for instance, that at the outset of flare, the aircraft is in trim and the pilot begins to pull back to reduce the sink rate. As the aircraft begins to enter ground effect the pilot in a conventional aircraft will tend to pull further. Thus, in landing a conventional aircraft without any trim adjustment, the pilot is holding back pressure on the column. If now, a corrective change is required in pitch attitude, the pilot accomplishes it either by further back pressure or a slight release of the back pressure. For the rate command/attitude hold type system, however, no back pressure is held. Consequently, if the attitude is to be reduced, the pilot must move the control forward from its neutral position. This feature of rate command/attitude hold systems has sometimes been remarked as undesirable.

From the above comments, it can be appreciated that a distinct tradeoff exists between the good features of aircraft which approach superaugmented configurations and a conventional statically stable aircraft as far as the trimability features are concerned. Some have "solved" this type of problem by using a lag lead for the augmentor equalization instead of the integrator lead combination of Figure 15 (References 4, 10, and 12) at the cost of retaining a long term divergence. Others have considered the augmentation of M_α or M_u instead of creating a pitch attitude related stability. As we shall see later in the next article, these can introduce unfavorable effects of a different kind and are not as straight forward in mechanization, especially in multiple redundant flight critical situations. The point of all of this is that a tradeoff in desirable flight stability and control features does appear to exist which was not contemplated in the original construction of the FARs. Existing research does not provide an unequivocal answer, but instead, indicates a tantalizing set of promises including, in this particular instance, a potential for safety enhancement (e.g. in windshears).

C. MECHANIZATIONAL SIDE EFFECTS ON FLYING QUALITIES OF HEAVILY AUGMENTED AIRCRAFT

In our consideration of relaxed static stability aircraft and the effects of heavy augmentation thereon, we have relied heavily on the generic RSS transport of Appendix A and the exemplary pitch rate command attitude hold command augmentation system of Figure 15. The question then naturally arises as to how representative are conclusions drawn from these examples to other cases. Specifically, will an augmentation system using a different system architecture result in significantly different conclusions and is the generic RSS transport aircraft, itself, reasonably representative?

The second question is easily answered because the aircraft characteristics are more or less a composite of a typical wide body high subsonic jet transport based on an elaborate set of studies from the NASA/Langley Energy Efficient Transport study. The primary issue that could be made relative to the aircraft-alone characteristics is the degree to which the static stability is relaxed. In our example, we have permitted negative static margins and have required that the control system cope with the resulting aperiodic aircraft-alone divergence. This was one of the characteristics which led to the assumption of an essential and hence multiply redundant flight control system. Heavy, or even superaugmentation naturally follows. The same end result can be reached from another direction where advanced technology flight control using fly-by-wire or fly-by-light concepts is the starting point. Again, essential configurations accomplished using multiple redundant control systems is a natural consequence and heavy augmentation is a reasonable follow-on. On the other hand, there are advantages to be gained in energy efficiency and performance that do not absolutely require operation of the aircraft alone in an unstable condition nor a multiple redundant fly-by-wire or fly-by-light flight control. Instead, the aircraft-alone is still configured as statically stable and a restricted authority augmentor is still applicable to improve the flying qualities. This option, which is consistent with a conservative approach to new

transport design is still sometimes referred to as relaxed static stability (e.g., References 11, 32). These only slightly relaxed static stability aircraft, of course, fall outside our scope. As noted at the outset of Section II, they do not fall into the category of heavily augmented aircraft with essential augmentation. They are mentioned again, here, primarily because of the semantics of "relaxed static stability" concepts. Our concern throughout has been on craft where this relaxation is sufficient to require a multiple redundant essential augmentation system. On this basis, the types of aircraft-alone instabilities and the aircraft-alone dynamics represented by the generic RSS transport aircraft of Appendix A are representative enough to permit the drawing of conclusions on a general basis.

The augmentation system, on the other hand, has many possible alternatives to accomplish at least some of the same ends. The alternative architectures, that is, feedback possibilities to accomplish desired heavy augmentation are many and diverse as discussed in connection with Table 1 of Section III. It was indicated there that two general effects were important. The first was to increase static stability and the second was to improve the short period damping. Those quantities useful for increasing the static stability fall into two fundamental categories. The first involve creation of an effective pitching moment proportional to an attitude quantity, such as the pitch attitude, θ, itself or an integral of pitching velocity, $\int q\,dt$. In this same class is the integral of normal acceleration because a_z is a linear function of the pitching velocity, q. With these systems, the effective aircraft dynamics include a new or created stability derivative, such as M_θ, which is not present in the conventional aircraft dynamics. The aircraft tends to be attitude stable rather than stable relative to angle of attack or speed. It, therefore, assumes a rigidity in pitch attitude rather than a stability relative to the air mass. These types of systems therefore provide an attitude hold feature in addition to stabilizing the divergence. At the same time, the speed stability for the primary pilot command is neutral as discussed in the last article.

In counterdistinction to the attitude type of system are those wherein an attempt is made to augment naturally occurring stability derivatives of the airplane alone for correction of a reduced static margin. This implies augmentation of M_{α} or M_u. In either of these cases, the speed stability will not be neutral. The nature of the speed stability, therefore serves as a fundamental distinction between systems. This is reflected in Tables 4 and 5. Those systems with neutral speed stability, that is systems based on attitude, pitch rate or normal acceleration, are assigned to Table 4, whereas those with non-neutral speed stability, based on angle of attack or speed, are listed in Table 5.

Other important distinction between possible systems are very much architectural dependent. These are considered side effects and can be more or less corrected by increasing the degree of complexity in the system design. They amount to those incidental features of a particular system mechanization which are over and above its primary purpose of improving static stability and short period damping. In the exemplary system, the primary side effect was the need to provide an up elevator compensation in turns proportion to $R_0 \tan \Phi_0$ to offset the steady state pitching velocity that occurs in turning. In Table 4 the exemplary system is the second one listed, $\int q\, dt$, $q \rightarrow \delta_e$. As noted previously, pitching velocity, $q \rightarrow \delta_e$, will go a long way toward improving the characteristics but will not completely reduce the divergence.

When other sensors, such as normal accelerometers, pitch gyros, etc. are used, the side effects may become more involved. They derive, in general, from three sources.

- <u>Biases associated with the particular instrumentation</u> used in the system. e.g., normal accelerometers pick up the total acceleration whereas the augmentation system ideally needs only acceleration perturbed from steady state conditions.

TABLE 4. SYSTEM ARCHITECTURAL POSSIBILITIES AND MECHANIZATIONAL SIDE EFFECTS FOR SUPERAUGMENTED AIRCRAFT

Systems Based on Attitude, Pitch Rate, or Normal Acceleration

$q \rightarrow \delta_e$

Reduces divergences, but does not get all the way to stability.
Requires some up-elevator relief in turns; e.g., $q_e = q - R_o \tan \Phi_o$

$\int q\, dt,\ q \rightarrow \delta_e$

Generally suitable for complete correction of instability.
Requires up-elevator relief in turns; e.g., $q_e = q - R_o \tan \Phi_o$

$\int a_z\, dt,\ G_{wo}q \rightarrow \delta_e$ (G_{wo} = Washout equalization)

Corrects for instability when operating on the frontside of the speed/power curves. Can have backside instability and equivalent backside in climbs.
Has bias ($a_{z_o} \neq 1$ g) when accelerometer is not oriented along stability axis for level flight; further bias in climbs and dives; yet another bias with a roll limit cycle.
Requires up-elevator relief in turns; e.g., $a_{z_e} = a_z - \cos \Theta_o \sec \Phi_o$ plus increment for q feedback in turn entry/exit.
Requires more airspeed compensation than attitude-based systems.

$1/(\hat{T}_{\theta_2}s + 1)\int Uq\, dt,\ G_{wo}q \rightarrow \delta_e$ [Pseudo a_z]

Generally suitable for complete correction of instability (replaces $d\gamma/dV$-based limitations with $1/T_{\theta_1}$; removes accelerometer bias issues).
Requires up-elevator relief in turns.
Requires more airspeed compensation than attitude-based systems.

$\theta,\ \dot{\theta} \rightarrow \delta_e$

Generally suitable for complete correction of instability.
Gain changes in turns, with associated F_s/g lightening, etc.
Requires elevator signal relief (trim) for $\theta \neq 0$.

$\theta,\ q$ or $\theta,\ G_{wo}q \rightarrow \delta_e$

Generally suitable for complete correction of instability.
Gain changes in climbing/diving turns.
Climb/dive steady-state signal relief.
Requires up-elevator relief in turn entries/exits, depending on specifics of G_{wo}.

TABLE 5. SYSTEM ARCHITECTURAL POSSIBILITIES AND MECHANIZATIONAL SIDE EFFECTS FOR SUPERAUGMENTED AIRCRAFT
Systems Based on Angle of Attack or Speed

<u>α_A, q or α_A, $G_{WO}q \to \delta_e$</u> (α_A = aerodynamic α)

Generally suitable for correction of instability.
Phugoid not much modified if G_{WO} focuses only on high frequencies.
Gust sensitivity associated with α_A.
α_{bias} position and scale factor errors (α sensor installation).
Requires trim set point.
Requires up-elevator relief in turn entries/exits, depending on specifics of G_{WO}.

<u>α_I, q or α_I, $G_{WO}q \to \delta_e$</u> (α_I = inertial α)

Generally suitable for correction of instability.
Phugoid not much modified if G_{WO} focuses only on high frequencies.
Requires trim set point.
Requires up-elevator relief in turns, depending on specifics of G_{WO}.

<u>Variants of α Systems</u>

$$\hat{\alpha} \doteq \frac{U_0}{[Z_w - M_w(Z_\delta/M_\delta)]} \frac{a_z}{U^2}$$

and other means of computing α.

<u>u_I, $G_{WO}q \to \delta_e$</u> (u_I = inertial u)

Generally suitable for correction of the instability.
May be subject to excessive pitching with a u_g input.
Must establish a set point or trim, $U = U_0$.
Phugoid damping ratio is reduced if G_{WO} focuses only on high frequencies.
Requires up-elevator relief in turns, depending on specifics of G_{WO}.

<u>u_A, $G_{WO}q \to \delta_e$</u>

As in item above.
Gust Sensitivity associated with u_A.
Scale and bias errors associated with u sensor installation.

- The degree of airspeed compensation for adjustment of the augmentor system total open loop gain. This differs with the nature of the sensor (e.g., a_z has a component $U_o q$ so normal accelerometer based systems will typically require a greater range of airspeed compensation than will θ or q based systems).

- The potential for correction of the aperiodic divergence is different for different feedback quantities (e.g., the a_z/δ_e airplane transfer function has a low frequency zero, $1/T_{h_1}$,* which can, itself, be negative. When this is the case, the divergence due to the negative static margin cannot be stabilized but simply approaches the value of $1/T_{h_1}$).

Table 4 summarizes these side effects for the attitude type neutral stability systems. The effects on flying qualities depend inherently on the degree to which these characteristics are corrected. Clearly, on a multiple redundant system, the complexity of correction is a major issue since any single channel should be made as simple and troublefree as possible. The issue for a given system then becomes how far one must go to correct the side effect created by the architectures selected. These are matters which have not been investigated on a comprehensive basis for the type of systems described in Table 4. At present, they would have to be considered on an ad hoc basis for each heavily augmented RSS transport design. The table in this sense, simply presents a checklist for particular design possibilities.

On the more general question of the broader applicability of results based on the exemplary system, such as those associated with T_q and T_{θ_2}, it can be stated that they are pertinent to the system possibilities of Table 4. The reason is that for all of these systems, the higher frequency properties approach those of the superaugmented ideal. Consequently, the issues drawn previously have a high degree of generality for systems covered by Table 4.

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*$1/T_{h_1} \doteq (1/3)(d\gamma/dV)$ when expressed in degrees/knot.

Relaxed static stability aircraft which are heavily augmented with systems based on angle of attack or speed to correct any static divergences have effective aircraft dynamic characteristics which are essentially conventional in form. This is particularly true as far as piloted control is concerned because the derivatives M_α or M_u for static stability correction are simply augmented to stabilizing levels. For aircraft responses to disturbances however, a distinction between conventional and heavily augmented aircraft may be pertinent depending upon the nature of the sensors used in the augmentation system. The disturbance sensitivities will specifically depend on whether an angle of attack system is based upon an inertial or aerodynamic angle of attack; similarly, for a speed system on whether inertial or air speed is used. The primary difference, however, between these types of systems and those based upon some form of attitude is in the nature of the stabilizing characteristic. The angle of attack system tends to stabilize the aircraft relative to the instantaneous (in the case of aerodynamic α_A) or steady state (for inertial $\alpha_I = W/U_0$) velocity vector orientation. This is, in essence, a weathercock stability and may involve significant pitching. The speed based systems create pitching moments proportional to changes from a trim or set speed U_0. There can be significant sensitivity to shears and forward gusts with this type of system since the aircraft must pitch to accomplish a balance of fore and aft forces.

Neither the angle of attack nor incremental speed feedbacks are especially simple to instrument, particularly on a multiple redundant basis. Systems of this type are more likely to involve sophisticated state reconstruction filters or observers and computation to generate the appropriate feedback signals. Unlike the attitude variety feedbacks, which do an excellent job in stabilizing the phugoid characteristics, angle of attack and speed are by themselves not particularly valuable in improving the phugoid dynamics. Indeed, in a normal airplane, the phugoid oscillation has very small angle of attack changes. The stability derivative, M_u tends to effect the phugoid frequency rather than its damping, which would require the creation of a new derivative, $M_{\dot{u}}$. This type of phugoid damping improvement, unfortunately, can create

some exciting pitching motions when the aircraft is disturbed by forward gusts or shears. Consequently, in both types of systems, a certain amount of pitching velocity or its equivalent is desirable at phugoid frequencies to improve the phugoid damping. These are indicated by the $G_{wo}q$ terms in Table 5, which signify a washed-out pitching velocity feedback or its equivalent. This type of feedback is, of course, also very effective for short period damping augmentation. When it is used for this purpose, with gains that are suitable for relatively heavy augmented aircraft, then the effective short period characteristics are dominated by the pitching velocity feedback. They can then be very similar to those of the attitude based systems as far as the short term time response characteristics are concerned.

The list of side effects for the angle of attack or speed base systems does not compare favorably with those for the attitude systems so heavily augmented aircraft using α or U as basic feedback quantities are probably not as likely as the rate-command/attitude-hold type system. This statement applies especially when the augmentor is at the essential level rather than on a dual or single thread non-flight critical basis.

Because pitching velocity feedbacks are likely to be present with relatively high gains in the Table 5 systems, the general issue of a distinction between an effective pitch attitude numerator lead, T_q, contrasting with the flight path lag, T_{θ_2}, present in the idealized superaugmented configuration is potentially present with these systems as well. Thus the conclusions previously drawn on this issue and those of the distinctions between ω_n and ζ with the conventional aircraft short period dynamics will apply. Thus, the general distinction between heavily augmented and conventional aircraft developed using the specific example of the Figure 15 exemplary augmentation system and the generic RSS transport aircraft will apply to some extent for all heavily augmented systems covered by Tables 4 and 5.

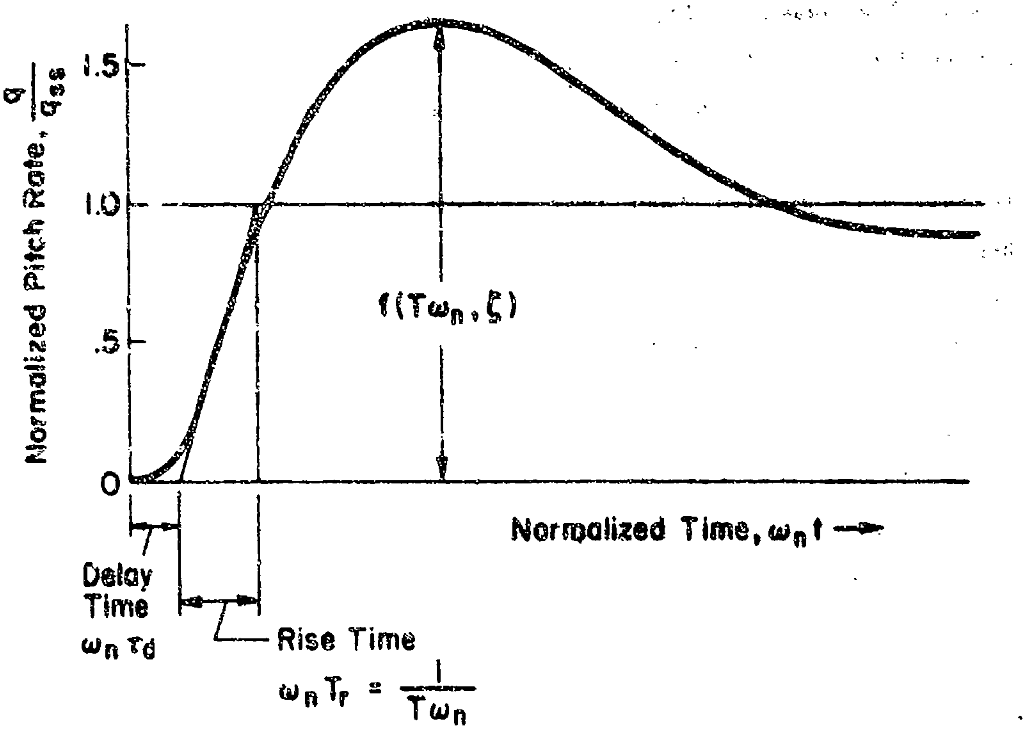
In the actual selection of an augmentation system architecture, the factors summarized in Tables 4 and 5 are among the major issues. However, in the design selection, the most important facets are their comparative simplicity as multiple-redundant system entities and their

inherent reliability and maintainability as single strings. In addition to the Table 4-5 factors, such as sensor biases and a need for correction of steady-state errors, additional considerations such as relative scale factors as installed, pickup of unwanted signals, vulnerability in normal operations and maintenance, ease of checking, etc. are also of major importance. When all of these factors are considered, in addition to those summarized, the rate-command/attitude-hold category represented in the Table 4 architectures will probably prevail in some form or other.

D. DELAY TIME INTRODUCED BY THE CONTROL SYSTEM

A common feature of all heavily augmented aircraft is the introduction by the control system of additional lags which may tend to delay the actual aircraft response buildup. The idealized response shown in Figure 8 starts with an instantaneous pitching acceleration for a step control surface input. In actuality, this response will appear more as shown in Figure 24. Because of the lags, after a step input is instantaneously applied there will be a very gradual buildup before the pitching acceleration begins to ramp off. This initial buildup delay is easily measured by a delay time τ_d. Although the discussion of this delay has been deferred to this point, it is the last item listed in the Table 2 comparison of pitch attitude response parameters for conventional and superaugmented aircraft.

In all the responses and example analyses described previously, the explicit assumption was that actuation sensing, filtering, and other real system lags were ignored. The delay time accounts for these. In the conventional aircraft, τ_d will be due primarily to the surface actuator and any additional lags that may be present in the manual control system. It, characteristically, is relatively small. Typical actuators, for example, appear like first order systems with time constants on the order of 0.05 sec or so for small amplitudes of movement. Thus, for a conventional airplane with a state of the art, fully powered, surface actuating system and manual control system, the effective time delay may typically be less than 1/10 sec.



General Form:

$$\frac{q}{q_{ss}} = \frac{(Ts+1)}{s\left[\left(\frac{s}{\omega_n}\right)^2 + \frac{2\zeta}{\omega_n}s + 1\right]}$$

T - Lead Time Constant

ω_n - Undamped Natural Frequency

ζ - Damping Ratio

Figure 24. General Pitch Rate Response to Step Pilot Input

With heavily augmented or superaugmented aircraft, additional considerations enter. As it turns out, all tend to increase the delay time. The baseline is, of course, the surface actuator shared with a conventional counterpart. The relaxed static stability aircraft which relies on the augmentation to restore favorable stability properties demands a controller-aircraft system minimum bandwidth which is characteristically greater than that present with a damping only augmentor. Because of this, the augmentation system bandwidth for rigid body control comes closer to intruding on the higher frequency flexible modes of the aircraft. So that these flexible modes do not become important in the aircraft/augmentor system stability, filters are often used in the control system to attenuate system signals which may arise due to the lower frequency lightly-damped flexible modes. This can be done using either low pass or notch filtering. Over the low frequency range associated with pilot control, either type of filter will appear as an effective lag.

In the augmentation system filters are sometimes required for certain sensors, such as normal accelerometers, to reduce unwanted inputs from the local vibratory environment. These filters also add to the net lag. Further, in the controller itself, a number of quite small time constant or pure time delay elements may be present. For instance, if the controller is digital, a pure time delay is introduced due to computing operations and filtering may be inserted for anti-aliasing at the input and smoothing digital to analog conversions at the output.

Finally, if the pilot command input is accomplished via a sidestick or low-force-input control column in a fly-by-wire (or light) controller installation, the manipulator will transmit both the desirable coherent pilot command signals and undesirable pilot-induced noise. The latter may be the random fluctuations in pilot control precision commonly referred to as remnant, or may be more excessive in vibratory environments. In either event, the pilot induced noise is ordinarily quite wide band, whereas the appropriate control signals are much narrower in frequency content. The manipulator signal accordingly may require filtering before it is presented to the flight controller.

All of these types of filtering and time delays are called out in the exemplary Figure 15 augmentation system. Their associated lags are individually quite small. However, unless great care is taken in the detailed design of the manipulator and other controller characteristics, they can add up to a sizable quantity. In fact, for the Approach and Land Test (ALT) version of the Space Shuttle Orbiter, such time delays approached 0.25 sec. This excessive delay pushed the ALT version of the Shuttle pitch rate response to or past the lower boundary of its own specification (Figure 20) which we have used herein as exemplary boundaries to provide a strawman frame of reference. As demonstrated in Reference 33 this effective delay in the ALT orbiter played an important role in the pilot induced oscillation encountered in Free Flight 5 during the approach and landing sequence. This and similar instances in advanced fighters, which are also RSS aircraft, have caused a great deal of emphasis to be placed on the effects of such delays.

While the flying qualities problems which may arise from excessive effective delay time are quite well understood, the quantitative picture is still clouded. The existing military specification, MIL-F-8785C (Reference 30) puts forth requirements for allowable time delays as a function of flying qualities levels. These are shown in Figure 25. It could be argued that Level 3 of the MIL-Spec borders on the unsafe and that the lower ranges of Level 2 require excessive pilot workload to accomplish even indifferent results and thus are marginally safe at best. The origins of the time delay requirements of Reference 30 are not well documented and these bounds were set up shortly after the basic problems were perceived to be a critical issue.

On the other hand, Reference 34 shows the boundaries from data developed for approaches and landing using the NASA DFRF F8 DFBW airplane. The high stress and low stress bounds depend on task precision. If Cooper-Harper ratings of 6-1/2 are taken as an upper limit, then the high stress boundary would indicate that delay times somewhat less than 0.22 sec might be permissible. On the other hand, it must be recognized that delay time cannot be viewed as an uncoupled entity. Instead, it is intrinsically connected with other flying quality metrics, such as rise

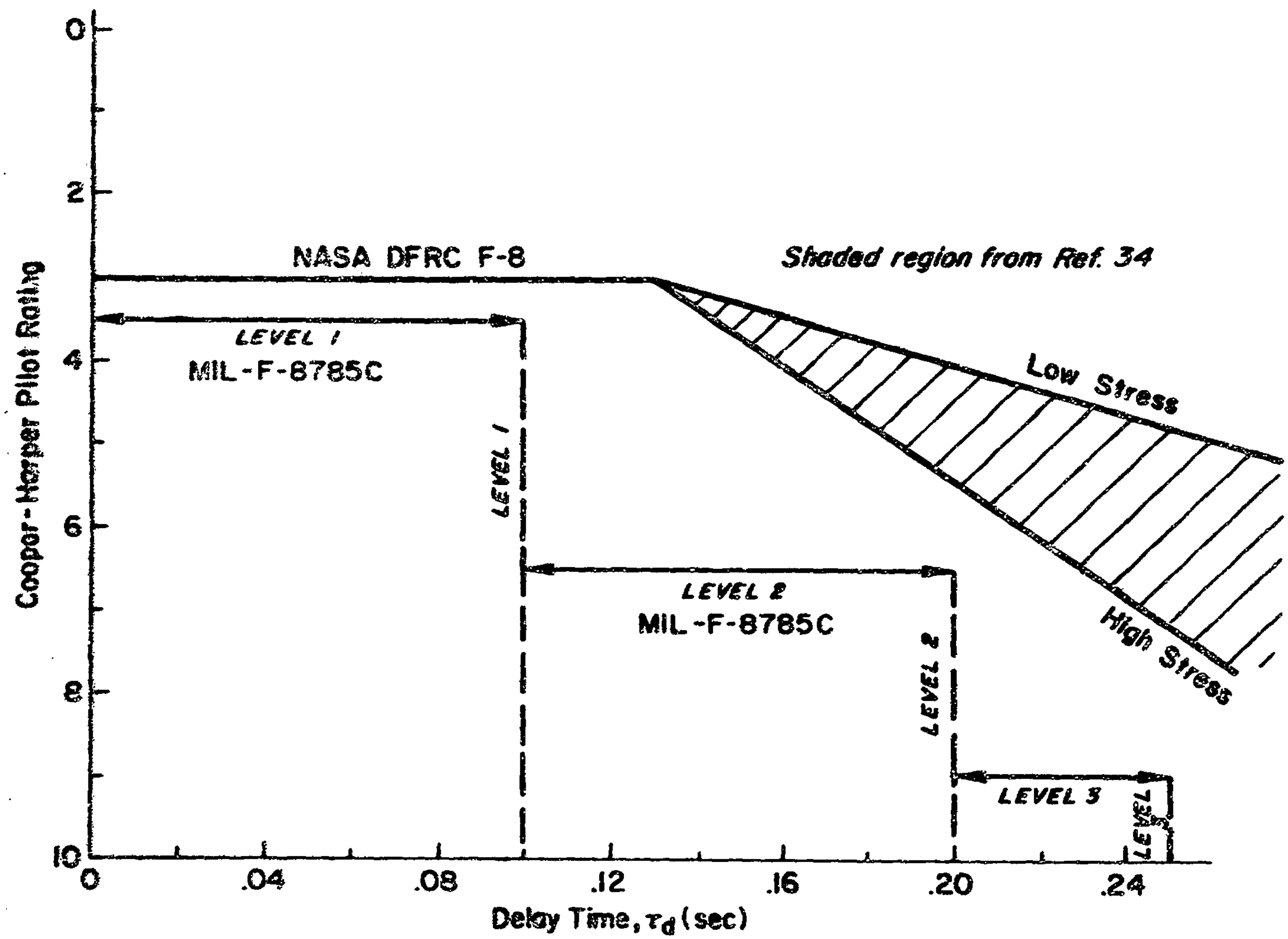


Figure 25. Comparison of MIL-F-8785C Time Delay Requirements with Data from Ref. 34

time. A system with a short rise time can surely accommodate more delay than a system with a longer rise time. Both time delay and rise time affect the attainable bandwidth of piloted control. This is recognized as one of the alternate specification factors in Reference 17.

Effective time delay, like most of the other issues raised in this study, is not specifically addressed in the FARs. It is unquestionably an important factor in the safety of operation of aircraft, especially in critical conditions which can be obtained during approach and landing and potentially other precision path control circumstances. The rise of heavily augmented aircraft makes effective time delays more important because of the tendency already cited for these aircraft to have values larger than on conventional airplanes. It is especially important in connection with delay time, as in most of the other flying quality issues brought up in this report, that precision control with high stress or high urgency conditions be emphasized from the standpoint of operational safety. A very large time delay could be accommodated if the aircraft is lined up well out on final approach and no urgent correction is required or extreme disturbance is present.

SECTION V

CONCLUSIONS

The primary thrust in this study has been to delineate and distinguish between the characteristics of so-called conventional aircraft and those aircraft of the future which may be equipped with high-authority essential stability augmentation systems. The key distinction sought had been those which might affect the flying qualities in precision high workload potentially critical conditions where flying qualities per se have their major impact on flight safety. To this end we have focused on those critical highest workload pilot/aircraft closed-loop operations which involve precision path control in the presence of unfavorable environmental conditions such as low visibility approaches and landings and turbulence and shear. The pilot's stress is thus at a high level, and pilot attention and gain levels will exhibit a comparable degree of urgency. Using tasks of this nature and an exemplary flight control system and generic RSS transport configuration, the following points have been demonstrated.

- Aircraft which are augmented at the essential level must be controllable throughout and after the failure transition(s). The aircraft-alone and transitional dynamics required in this connection are not currently well defined. An oversimplified and not too well supported criterion requiring any aircraft divergence to have time-to-double amplitudes greater than 6 sec provides a stop gap value. Additional research, which includes crew functions and behavior in takeover and recovery during the failure transitional phase, is badly needed.

- The path/attitude lag, measured by the aircraft time constant T_{θ_2}, is the same for both heavily augmented and conventional aircraft (only elevator control is present).

- The pitch attitude characteristics are important both for their own sake in attitude control tasks and as an adjunct inner loop in path control tasks.

- The effective pitch attitude dynamics for a conventional and for heavily or superaugmented aircraft are similar in dynamic form.

- The governing parameters in short term pitch attitude response for a conventional aircraft are fundamentally dependent on the airplane lift curve slope, static margin, etc., whereas the governing parameters for a heavily augmented aircraft depend on control system and aircraft surface effectiveness quantities. Thus while the short term pitch transfer function characterstics have the same form, the quantities within this form can be drastically different, both in kind and in quantitative degree.

- Existing flying quality data and criteria based on conventional aircraft may not be directly applicable to superaugmented aircraft and only partially applicable at best to heavily augmented aircraft. Thus, flying quality considerations based on many years of experience, much data and attempts to develop understanding, cannot necessarily be used by analogy for heavily augmented future aircraft.

- The experimental data base for highly augmented aircraft is very sparse. Applicable data that do exist indicate that superaugmented and, heavily augmented aircraft can exhibit good flying qualities and hence safe operations.

- The FARs were constructed in an era where the distinctions drawn in this study between heavily augmented and conventional aircraft were unknown. With some special interpretations and/or modifications, the FARs can be adjusted to accommodate the new technologies and new flight control forms.

- There exists insufficient data at present to resolve flying qualities and safety issues associated with
  - Aircraft pitch attitude lead, T_q, different from flight path lag T_{θ_2}
  - Short period dynamics independent of static margin
  - Neutral speed stability in pitch rate command/attitude hold systems
  - Larger effective time delays in the control system

- The detailed mechanizational nature of the augmentation system introduces specific side effects which have a potential impact on piloted control and augmentation system complexity. Very little data exists on the flying qualities and safety importance of particular side effects, and on the tradeoff between elimination of side effects and system complexity.

In general, the primary conclusion of this study is that heavily augmented aircraft have flight characteristics which differ in several important ways from current conventional airplanes. These flying qualities differ in kind and degree and in some respects are not compatible with a rigid construction of the current FARs. On the other hand, the advanced systems themselves offer promise to enhance rather than reduce safety. At present, insufficient data exists to resolve all the questions raised here. Instead, the important distinctions have been developed, the key issues identified, and the governing parameters delineated. This should serve as the basis for

- A review and consideration of potential modifications to the FARs for Part 25 aircraft.
- Outline of an interim flight test guide for heavily-augmented and superaugmented aircraft
- A point of departure for both generalized research to resolve the issues raised and ad hoc considerations pertinent to specific flight control/RSS aircraft flying qualities and safety considerations

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APPENDIX A

GENERIC RSS TRANSPORT AIRCRAFT

A model of a hypothetical high subsonic jet transport with relaxed static stability was generated and used as a basis for some of the preceding analysis. This generic transport was based in part on two RSS designs from the NASA/Langley Energy Efficient Transport study.

- The Lockheed L-1011-RE (Ref. 11), which has a 38 percent reduction in horizontal tail area compared to the conventional L-1011. This design is considered ready for flight test.

- The Boeing IAAC, (Ref. 12), design in which the wing is further forward on the fuselage and the horizontal tail area is reduced 42 percent compared to a conventional baseline aircraft of comparable specifications.

The generic RSS transport developed here is a wide-body high subsonic jet transport with an all-moving horizontal tail. Data were generated for two flight conditions: a) Sea level approach at 1.3 V_{stall}; and b) 38,000 ft cruise at Mach 0.74. The horizontal tail area was reduced 40 percent with respect to a reference conventional design with the tail length held fixed. The aerodynamic coefficients were corrected for the tail area reduction. $C_{M_{\dot{\alpha}}}$, C_{M_q}, and $C_{M_{\delta_H}}$ were assumed dominated by the tail and reduced 40 percent compared to the conventional baseline. Small tail corrections were made in the lift derivatives, but effects on drag stability derivatives were neglected. No weight changes were made, and the effects of c.g. location on the derivatives were neglected except for C_{M_α}. Transfer functions for δ_H, u_g, and w_g including coupling numerators, were generated from the STI Airframe Transfer Function program (ATFP) for approach and cruise at C_{M_α} values corresponding to +5, 0 and −5 $\%\bar{c}$ static margins. The model parameters are tabulated in Table A-1 and the δ_H transfer functions are tabulated in Table A-2.

TABLE A-1. GENERIC RSS TRANSPORT MODEL PARAMETERS

PARAMETER		FLIGHT CONDITION APPROACH	FLIGHT CONDITION CRUISE	STATIC MARGIN
c	ft	24	24	
s	ft^2	3,460	3,460	
U_o	fps	230	230	
h	ft	0	38,000	
W	[illegible]	300,000	300,000	
I_y	slug-ft^2	12×10^6	12×10^6	
C_L		1.38	0.523	
C_{L_α}	rad^{-1}	5.88	5.16	
$C_{L_{\dot\alpha}}$	rad^{-1}	-3.30	-3.80	
C_{M_q}	rad^{-1}	-10.6	-10.5	
C_D		0.21	0.0355	
C_{D_α}	rad^{-1}	0.842	0.573	
$C_{D_{\delta_H}}$	rad^{-1}	0	0	
$C_{L_{\delta_H}}$	rad^{-1}	0.96	0.82	

TABLE A-1. (CONCLUDED)

PARAMETER		FLIGHT CONDITION APPROACH	CRUISE	STATIC MARGIN
$C_{M_{\delta_H}}$	rad^{-1}	-2.30	-1.69	
C_{M_α}	rad^{-1}	-.294 0 .294	-.258 0 .258	+5% $\bar{c}$ 0 -5% $\bar{c}$
X_u	sec^{-1}	-0.0427	-0.00749	
X_w	sec^{-1}	0.0546	-0.00429	
Z_u	sec^{-1}	-0.280	-0.1029	
$Z_{\dot{w}}$	sec^{-1}	0	0	
Z_w	sec^{-1}	-0.619	-0.446	
M_u	$(\text{ft-sec})^{-1}$	0	0	
$M_{\dot{w}}$	$(\text{ft-sec})^{-1}$	-0.000326	-0.000102	
M_q	sec^{-1}	-0.241	-0.202	
X_{δ_H}	ft/rad sec^2	0	0	
Z_{δ_H}	ft/rad sec^2	-22.4	-50.4	
M_{δ_H}	$1/\text{rad sec}^2$	-1.00	-1.94	
M_α	sec^{-2}	-0.128 0 0.128	-0.296 0 0.296	+5% $\bar{c}$ 0 -5% $\bar{c}$

TABLE A-2. GENERIC RSS TRANSPORT HORIZONTAL TAIL TRANSFER FUNCTIONS
(Listed in Order of Static Margin: 5%, 0, $-5\%\bar{c}$)
Shorthand notation: $(a)[\zeta,\omega] = (s + a)[s^2 + 2\zeta\omega s + \omega^2]$

	APPROACH	CRUISE
Δ	[0.0334, 0.1278] [0.873, 0.555] (0) (0.718) [0.964, 0.1345] (−0.1337) (0.909) [0.496, 0.203]	[0.01219, 0.01219, 0.0595] (0.) (0.01056) (0.1553) (0.562) (0.238) (0.941) [0.1592, 0.00781]
$N^u_{\delta_H}$ fps/rad	−1.225 (0.962) (−16.58) −1.225 (0.981) (−16.60) −1.225 (1.00) (−16.62)	0.216 (0.398) (315) 0.216 (0.408) (315) 0.216 (0.419) (315)
$N^w_{\delta_H}$ fps/rad	−22.4 (10.52) [0.0996, 0.1958] −22.4 (10.52) [0.0996, 0.1958] −22.4 (10.52) [0.0996, 0.1958]	−50.4 (27.7) [0.0540, 0.0678] −50.4 (27.7) [0.0540, 0.0678] −50.4 (27.7) [0.0540, 0.0678]
$N^\theta_{\delta_H}$ rad/rad	−0.995 (0.0713) (0.582) −0.995 (0.0706) (0.595) −0.995 (0.0699) (0.608)	−1.931 (0.00646) (0.437) −1.931 (0.00648) (0.448) −1.931 (0.00651) (0.459)
$N^{\dot{h}}_{\delta_H}$ fps/rad	22.4 (0.00318) (−2.32) (2.68) 22.4 (0.00398) (−2.35) (2.70) 22.4 (0.00474) (−2.37) (2.73)	50.4 (−0.00416) (−3.32) (3.61) 50.4 (−0.00388) (−3.36) (3.65) 50.4 (−0.00361) (−3.40) (3.69)
$\frac{u}{\delta_H} = \frac{N^u_{\delta_H}}{\Delta}$, etc.		

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